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Date: 30 June 1965

Volume 1

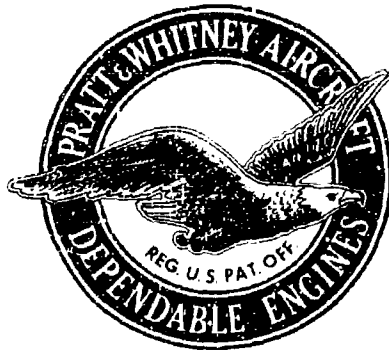
## SUPERSONIC TRANSPORT AIRCRAFT ENGINE

### PHASE D-B DEVELOPMENT PROGRAM

#### FINAL REPORT (U)

Prepared Under Contract FA-SS-65-18

Period Covered 1 January through 30 June 1965



PWA-E. H. Document Control  
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PWA-2600

## FOREWORD

This report describes the work that was accomplished by Pratt & Whitney Aircraft during the period 1 January 1965 through 30 June 1965 in accordance with the requirements of contract FA-SS-65-18 entitled "Development of Supersonic Transport Engine - Phase II-B". The report is submitted to fulfill the requirements of Item 7, Section D of the contract work statement.

This report is classified as CONFIDENTIAL in compliance with the provisions of DD Form 254 dated 1 January 1965 provided for this contract.

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## **INTRODUCTION**

The overall objectives of the program conducted under contract FA-SS-65-18 were to continue: the design of the STF 219 duct-heating turbofan engine, liaison with aircraft manufacturers to ensure optimum engine and ejector-reverser installation, and verification of major component performance by approximately full-scale component testing. This program was a continuation of the contractor's design and test effort on supersonic transport powerplants and was aimed at achieving further advances in engine design and component state-of-the-art over those submitted in the Phase II-A proposal for the supersonic transport engine.

In accordance with the requirements of contract FA-SS-65-18, the program was divided into 15 major fields of effort corresponding to the tasks listed in Section B of the contract work statement. These fields of effort included engine design in addition to research and development on compressors, primary combustion, turbines, augmentors, inlet and exhaust systems, noise reduction, controls and accessories, bearings and seals, and fuels and lubricants. Also, investigations were conducted on installation optimization, materials and manufacturing techniques, and supporting design considerations such as maintainability, reliability, and value engineering. A discussion of the work accomplished in each of these fields is presented in separate sections of this report in an order corresponding to the work statement items of the contract.

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ITEM 1

ITEM 1  
INSTALLATION COORDINATION



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## ITEM 1 - INSTALLATION COORDINATION

## OBJECTIVE

The Contractor continued to work with the airframe contractors to ensure optimum installation arrangements for the engine and ejector reverser in the airframe. Engine cycle studies were also included in this work.

A. INSTALLATION COORDINATION MEETINGS  
WITH AIRPLANE COMPANIES1. INSTALLATION COORDINATION WITH THE BOEING AIRPLANE  
COMPANYa. Introduction

Six coordination meetings were held between the Contractor and the Boeing Airplane Company during Phase IIB. The most significant topics discussed in the course of these meetings are summarized below:

- Performance. IBM performance decks for both turbofan and turbojet were given to Boeing. Changes were incorporated in these programs, at Boeing's request, to enable more efficient use to be made of their computer time.
- Ejector concepts. Improved mechanical and aerodynamic ejector concepts were studied, the most recent of which is a sliding shroud ejector (a fixed shroud ejector was used during Phase IIA).
- Noise. Information was exchanged several times, and Boeing witnessed a full-scale noise test in which a J57 afterburning engine, using a "boiler plate" SST type ejector, was run.
- Engine-to-inlet compatibility. Various constants for the Contractor's analog computer program for studying engine-to-inlet compatibility were supplied to Boeing for both the turbofan and turbojet.

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- Turbine inlet temperature. The Contractor's turbine program was reviewed several times during Phase IIB. Data, hardware, and future planning were discussed.
- Weight. Weights were updated periodically. Weights for a specific installation were supplied to Boeing with each installation drawing.
- Engine-to-wing mating. A reduction in the base drag in the region between the engine and the wing at the wing trailing edge was made. Compatibility with Boeing's latest installation requirements was achieved.

The above items, with the exception of engine-to-wing mating, are covered in detail elsewhere in this report. This section of the report will, therefore, be devoted to a discussion of the engine-to-wing mating work done during Phase IIB.

**b. Summary of Engine-to-Wing Mating Work**

A reduction in base drag was accomplished by changing the shape of the ejector from round to octagonal, and by moving the ejector cant point aft relative to its Phase IIA position.

Figure 1-1 shows a comparison between the Phase IIA and Phase IIB configurations. Figure 1-2 shows the benefits obtained by moving the ejector cant point aft.

**c. Nomenclature**

The following nomenclature is used in describing the engine-to-wing mating work.

- Equivalent diameter. An octagonal ejector is used to eliminate base drag. As it is cumbersome to describe the size of an octagon, i.e., by dimensions across flats and corners, the terminology "equivalent diameter" or "equivalent round" has been adopted. This expression refers to the diameter of a round ejector which has the same geometric area as the particular octagonal ejector under discussion.
- Cant point. For all intents and purposes, the plane through which the ejector is canted intersects the engine center line at a point about which the ejector is a body of revolution. This point is called the ejector cant point. The location of

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the cant point influences the following:

- Ejector cant angle
- Position of the ejector relative to the wing.

#### d. Background

##### (1) Selection of Cant Point Location

For a given size engine and ejector combination, and for a given exhaust gas target point, it is advantageous to move the cant point aft. As the cant point moves aft, the ejector cant angle decreases while the ejector stays close to the trailing edge of the wing with very little interruption of internal wing structure. This is illustrated in Figure 1-2. It is desirable to have a decreased cant angle because of ejector internal aerodynamic and mechanical reasons. It is also desirable to have the ejector close to the wing because of the consequent reduction in base drag, and simpler wing mating.

In the course of studying how to obtain these desirable features, the ejector was canted in three different places. In Phase IIA the ejector was canted at the rear mount plane. Early in Phase IIB, the cant point was moved aft to the rear face of the turbofan primary nozzle to determine the effect on base drag and ejector cant angle. Later in Phase IIB, the cant point was moved slightly forward to a plane which passes through the throat of the fan nozzle (or the afterburner nozzle in a turbojet). This latter position assured symmetry downstream of the nozzle choke point. Boeing selected this cant point as the one which best suited their installation. In each of the foregoing cases, however, the canting results in some non-symmetry upstream of the cant point which will impose additional mechanical and aerodynamic complexity.

##### (2) Mounts, Tailflaps, and Inlet Extensions

As the installation progressed, the engine (and mounts) were moved rearward to reduce base drag. Finally, in order to get the variable exhaust nozzle (tailflaps) entirely out from under the wing where it could adversely effect base drag, the ejector was positioned relative to the wing such that the hinge point of the tailflaps was either in line with or aft of the wing trailing edge. Since the Boeing inlet position was fixed relative to the wing, the engine had to fill the space between the ejector (as positioned by the tailflap hinge point) and the inlet. The shorter turbofan required an extension or spacer between the engine inlet and the Boeing inlet, see Figure 1-1. The longer turbojets with full afterburners did not require an inlet extension and, in some cases, they were long enough to necessitate moving the tailflap hinge point aft of the wing trailing edge.

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### (3) Boeing's Installation Requirements

The differences between the installation drawings presented later in this section reflect the changes in Boeing's installation requirements as well as changes to the engine. As detailed design studies progressed, various aspects of the powerplant, including length and diameter, were revised. Close coordination with Boeing was essential to ensure that these revisions were compatible with the frequently changing airframe requirements.

The installation drawings were based on the latest Boeing requirements in the following areas:

- Wing contour. Wing cross section at the outbound nacelle location was provided by Boeing.
- Inlet position relative to the wing. This was changed as a function of the inlet flow field.
- Exhaust gas target point. This was fixed to be compatible with good cruise performance and by the location of the horizontal stabilizer.
- Permissible limits of mount locations. These limits were supplied by Boeing from time-to-time based on the latest wing configuration.
- Thrust reverser targeting requirements. Reverse thrust requirements and the possibility of re-ingestion were taken into consideration.
- Ejector position relative to the wing at the wing trailing edge. This position was varied from time to time depending on its influence on the favorable interference effect between the engine and the wing.

As a rule, the inlet, the exhaust gas target point, and the position of the ejector variable tailflap hinge point were all fixed relative to the wing. The objective was to fit the engine to these points using extensions, canting, etc., in such a manner as to result in the minimum amount of interruption of wing internal structure.

### (4) Ejector Concepts

Improved aerodynamic and mechanical concepts were studied during

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Phase IIB. These studies resulted in the sliding shroud ejector (a fixed shroud ejector was used in Phase IIA). The sliding shroud ejector uses more efficiently the energy available from the engine and secondary air streams, and reduces the minimum wrap of the ejector around the engine. The sliding shroud ejector also has a higher L/D than the fixed shroud ejector. Its longer length for a given diameter, together with the translating shroud capability, has the potential for more flexibility in accommodating thrust reverser targeting. This ejector appears on the latest turbojet and turbofan installations.

#### (5) Afterburner Concepts

The afterburner for the turbojets progressed from a short, partial (acceleration) type unit with a Mach 2.0 limitation to a longer, full afterburner with Mach 2.7 capability (Boeing indicated that they required a Mach 2.7 or full afterburning capability). This primarily accounts for the increased length of the turbojet engines toward the latter part of Phase IIB.

#### (6) Flow Schedules and Turbine Inlet Temperature

Above Mach 2.0 on both the turbofan and the turbojet, the corrected engine airflow as a function of Mach number may be selected over a range of values. This flexibility gives Boeing an opportunity to choose the airflow schedule which results in the best match between engine and inlet. A "high", "base" and "low" flow schedule were offered. In Phase IIA, Boeing selected a base flow turbofan. In order to provide an opportunity for comparison of turbojets, all three flow schedules were offered in the latter part of this Phase. For each flow schedule, configurations for both the 2000°F and 2300°F turbine inlet temperatures were presented. Boeing could, therefore, examine the trades involved in starting initial service at the lower temperature with subsequent growth to the higher temperature within the same external envelope.

#### e. Discussion of STF 219B Turbofan Installations

The study of new ejector and installation concepts began with the Phase IIA configuration, (see Figure 1-3 ), which was a 600 lb/sec engine with a cylindrical ejector of 76.00 inch diameter. This arrangement produced a base drag area between the ejector and the wing trailing edge of approximately 425 sq. in.

The object of the new studies was to reduce this base drag area, thereby improving on the Phase IIA installation. Wing contour, mount location,

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and exhaust gas targeting information, as well as nacelle capture area and a 5° inlet angle relative to the horizontal reference datum, was supplied by Boeing and was used as the basis of these studies. An octagonal ejector was selected to further reduce the base drag area at the wing trailing edge. The flat side on top of the octagonal ejector offered possibilities of closer coupling to the wing surface than the circular section of the Phase IIA ejector.

A number of new arrangements of the engine and the octagonal ejector relative to the wing were investigated. These arrangements are summarized below.

- Figure 1-3 (Phase IIB configuration) shows how an octagonal ejector with the same equivalent area as the 76.00 inch diameter ejector of Phase IIA reduces the base drag area to 398 sq. in.
- As in Figure 1-3, Figure 1-4 shows an arrangement with the engine and ejector tangent to the wing. The rear engine mount plane was moved aft 40.00 inches (relative to its Phase IIA position) which required a 33.50 inch extension to the engine inlet case. The ejector was canted at an angle of 5° at the rear mount plane. The base drag area was reduced to 337 sq. in.
- Figure 1-5 shows an arrangement with the rear engine mount plane moved aft 30.00 inches. The ejector was canted 8° at the rear mount plane and the engine/ejector was inserted 4.00 inches into the wing. A 31.00 inch extension was required. The base drag area was reduced to 320 sq. in.
- Figure 1-6 shows an arrangement identical to Figure 1-5 except that the ejector cant point was moved aft to the primary nozzle plane. The cant angle is 6°. A 40.00 inch extension was required for this. The base drag area was reduced to 230 sq. in.
- Figure 1-7 shows an arrangement with the rear mount plane moved aft 40.00 inches and the ejector canted at 8° at the primary nozzle plane. The engine/ejector was inserted 4.00 inches into the wing. A 40.00 inch extension was required. The base drag area was reduced to 150 sq. in.

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- Figure 1-8 shows the best arrangement from a base drag reduction standpoint. The ejector tailflap hinge line was moved aft to the wing trailing edge and set tangent to it. The ejector was canted  $9^\circ$  at the primary nozzle plane. This configuration required a 51.50 inch extension, but reduced the base drag area to zero.

At this point in the program, Boeing specified a larger ejector diameter. Their area ruling dictated an ejector area/inlet area ratio = 1.7. Based on this, the ejector equivalent diameter was increased to 78.00 inches. Figure 1-9 shows an arrangement with this larger ejector canted  $11^\circ 30'$  at the rear mount plane. The ejector tailflap hinge line was aligned with and set tangent to the wing trailing edge. This configuration required a 44.00 inch extension. The base drag remained at zero.

The ejector cant point was moved 10.00 inches forward of the primary nozzle plane to the throat of the fan nozzle. This preserved symmetry downstream of the fan nozzle choke point. Figure 1-10 shows an arrangement using this new cant point on the new 78.00 inch diameter equivalent ejector. The ejector tailflap hinge line was aligned and set tangent to the wing trailing edge. The ejector cant angle was  $8^\circ 45'$ . This required a 49.00 inch extension. Here again the base drag area was zero.

Figure 1-11 shows an arrangement for a 650 lb/sec engine with an 81.00 inch equivalent diameter ejector. The ejector is positioned so that the tailflap hinge line is aligned with the wing trailing edge. The upper surface of the octagonal ejector is set 3.00 inches above the wing upper surface at the wing trailing edge as requested by Boeing. The nacelle inlet capture area center point location of 29.60 inches down from the horizontal reference line was changed to 28.30 inches by Boeing. The ejector cant angle became  $8^\circ 41.7'$  and a 17.4 inch extension was required.

Figure 1-12 shows a further Boeing revision. The inlet angle was reduced to  $2^\circ 15'$  from  $5^\circ$ . Figure 1-12 is identical to Figure 1-11 except for the aforementioned inlet angle and the ejector cant angle which was increased to  $11^\circ 2.1'$ . A 30.63 inch extension was required.

Figure 1-13 shows an arrangement for a 600 lb/sec engine/ejector system and is similar to Figure 1-10 except that it was laid out to the latest Boeing geometry. The latest Boeing inlet centerline location of 27.30 inches down from the horizontal reference datum was used. The ejector was positioned as before, 3.00 inches above the wing with the tailflap hinge line aligned with the wing trailing edge. The octagonal

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ejector has a 78.00 inch equivalent diameter and was canted 10.00 inches forward of the primary nozzle plane. The ejector cant angle is  $10^{\circ} 55'$  and the configuration required a 36.98 inch extension.

The previous installation studies all involved fixed shroud ejector designs. A sliding shroud ejector was evolved in Phase IIB. This shroud makes more effective use of the energy in the engine and secondary air streams, reduces the minimum wrap of the ejector around the engine, and provides greater reverser targeting flexibility. On the turbofan, the shroud translates to three positions. The cruise position (blow-in doors closed) is the forward-most position. For take-off and up through the blow-in-door operating range, the shroud translates somewhat rearward relative to the cruise position. The shroud translates further rearward for reverse. The sliding shroud ejector is described in detail elsewhere in this report. Figure 1-14 shows a turbofan configuration which incorporates a sliding shroud ejector. Figure 1-14 also shows the most recent STF 219B accessory arrangement.

f. Discussion of STJ 227B Turbojet Installations

A 500 lb/sec afterburning turbojet engine installation was presented to Boeing for their initial Phase IIB studies. A number of new arrangements of the STJ 227B engine/ejector relative to the wing were investigated. The arrangements are summarized below. Two maximum turbine inlet temperatures were considered:  $2000^{\circ}\text{F}$  and  $2300^{\circ}\text{F}$ .

- Figure 1-15 shows an arrangement using an octagonal ejector of 75.00 inch equivalent diameter on a  $2300^{\circ}\text{F}$  turbine inlet temperature engine with partial augmentation. The inlet cowl was located 27.00 inches down from the horizontal reference line. Wing contour, reverser targeting, and the  $5^{\circ}$  inlet angle were supplied by Boeing and were used as the basis for these studies. The ejector was positioned so that the tailflap hinge line was aligned to and set tangent with the wing trailing edge. The ejector was canted at the rear engine mount plane with a  $10^{\circ} 15'$  angle required to hit Boeing's exhaust gas target point. This configuration required a 13.40 inch extension at the engine inlet.
- Figure 1-16 shows an arrangement identical to Figure 1-15 except that the ejector cant point was moved aft to the throat of the afterburner nozzle, which reduced the cant angle to  $7^{\circ} 40'$ . This required a 13.80 inch extension.

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The 2000°F maximum turbine inlet temperature engine incorporated a full afterburner from its inception and was thus longer than the 2300°F engine with partial augmentation. An 81.00 inch equivalent diameter ejector was used. Figure 1-17 shows an arrangement using this size octagonal ejector on a 2000°F turbine inlet temperature engine. The ejector, canted at the rear engine mount plane, required a cant angle of 8° 25'. The overall length of the engine and ejector eliminated the need for an inlet extension.

Figure 1-18 shows an arrangement identical to Figure 1-17 except that the ejector was canted at the throat of the afterburner nozzle, which reduced the required cant angle to 6° 30'.

At this point in the program the engine size was increased to 525 lb/sec. Figure 1-19 shows an arrangement of this engine with a partial afterburner and its required 76.8 inch equivalent diameter ejector for turbine inlet temperature of 2300°F. This configuration was positioned relative to the wing according to the latest Boeing data. A capture diameter of 56.18 inches located 27.75 inches down from the horizontal reference line with an inlet angle of 2° 15' was used. The ejector tail-flap hinge line was aligned with the wing trailing edge and with the upper surface of the octagonal ejector 3.00 inches above the wing upper surface at the wing trailing edge. The ejector was canted slightly forward of the afterburner nozzle throat at 10° 50'. A 21.00 inch inlet extension was required.

Figure 1-20 shows the corresponding arrangement for a fully augmented 2000°F engine with its required 83.00 inch equivalent diameter ejector. The engine was positioned relative to the wing similar to that shown in Figure 1-19 except that the length of the engine moved the ejector tail-flap hinge line 35.00 inches aft of the wing trailing edge. The ejector was canted at a point 2.40 inches forward of the afterburner nozzle throat at 8°. No extension was required.

The previous installations all had fixed shroud ejectors. Later installations used the sliding shroud ejector. On the turbojet, the shroud translates such that it has one position for forward flight, and another for reverse. This ejector is described in detail elsewhere in this report. Six full afterburning turbojet configurations incorporating the sliding shroud ejector were presented to Boeing. These six configurations comprised a high, base, and low flow version of the 2000°F and the 2300°F engine. Each configuration was adjusted to its particular flow schedule and temperature for comparative purposes. As in the case of the turbofan, once the comparison has been completed and a selection made, one configuration may then evolve which permits growth from

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2000 °F to 2300 °F within the same external envelope. The six configurations are summarized below:

- Figure 1-21 shows an installation study for a 2000 °F turbine inlet temperature, high flow engine. The ejector size used for this configuration was 78.00 inch diameter equivalent and was canted 1.50 inches forward of the afterburner nozzle throat at 7° 55'.
- Figure 1-22 shows an installation study for a 2000 °F turbine inlet temperature, base flow engine. The ejector size used for this configuration was 75.00 inch diameter equivalent and was canted 1.80 inches forward of the afterburner nozzle throat at 8° 35'.
- Figure 1-23 shows an installation study for a 2000 °F turbine inlet temperature, low flow engine. The ejector size used for this configuration was 72.00 inch equivalent diameter and was canted 1.75 inches forward of the afterburner nozzle throat at 8° 50'.
- Figure 1-24 shows an installation study for a 2300 °F turbine inlet temperature, high flow engine. The ejector size used for this configuration was 78.00 inch equivalent diameter and was canted 1.65 inches forward of the afterburner nozzle throat at 8° 9'.
- Figure 1-25 shows an installation study for a 2300 °F turbine inlet temperature, base flow engine. The ejector size used for this configuration was 75.00 inch equivalent diameter and was canted 1.80 inches forward of the afterburner nozzle throat at 8° 35'.
- Figure 1-26 shows an installation study for a 2300 °F turbine inlet temperature, low flow engine. The ejector size used for this configuration was 70.00 inch equivalent diameter and was canted 2.00 inches forward of the afterburner nozzle throat at 9° 45'.

It should be noted that in the above six installation studies the overall length of the engine/ejector eliminated the need for an inlet extension.

Figure 1-27 shows the most recent SIJ 227 accessory arrangement.

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### a. Introduction

- Engine accessories. The Phase IIA arrangement was revised in order to relocate the accessories away from the bottom of the engine.
- Ejector. An octagonal blow-in-door ejector was studied as an alternative to the 12-sided design presented in Phase IIA.
- Engine cant and wing relationship. A study was made to determine the most desirable location for canting the engine as the result of Lockheed's request that the engine be canted  $4^{\circ}$  in a downward direction.
- Sonic boom. The size of the engine was changed in the light of the FAA's reduction in the sonic boom overpressure requirements.
- Turbojet installations. A number of installation sketches was prepared for various versions of the turbojet, which was reintroduced during Phase IIB.

Pratt & Whitney Aircraft restudied the accessory arrangement proposed in Phase IIA. This arrangement consisted of engine accessories driven by a gearbox located on the bottom of the engine and a power take-off gearbox located on top of the engine which supplied power to drive the airframe accessories. Lockheed requested that all engine accessories containing combustible fluids be moved from the bottom of the engine to reduce the danger of fire in the event of a collapsed landing gear or belly landing.

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were removed from the bottom. This arrangement was rejected by Lockheed because the accessories in the upper quadrants interfered with their front engine mount structure. The arrangement also prevented ready access to the power take-off gearbox.

Continued study rejected the use of two separate towershafts and gearboxes for engine accessories because of internal gearing difficulties. A new arrangement (Figure 1-29) was devised with the engine-driven accessories located on the left side of the engine approximately  $45^\circ$  below the engine horizontal center line. The remaining accessories were located at the same level on the opposite side of the engine. Although this arrangement involved greater difficulty in removing the accessories from the bottom of the engine, the arrangement was generally acceptable for the 700 lb/sec engine size. When the size was reduced to 650 lb/sec this accessory arrangement became less attractive, as the units were crowded together in the reduced circumferential space. An alternate arrangement, Figure 1-30 evolved locating the engine accessory drive shaft on the right horizontal centerline, with the accessories grouped differently from Figure 1-28.

#### c. Octagonal Ejector

Early in Phase IIB Pratt & Whitney Aircraft studied the use of an octagonal blow-in-door ejector as an alternate to the circular door 12-sided design presented in Phase IIA. Figure 1-31 is a schematic drawing of the octagonal ejector sent to Lockheed. The octagonal ejector dominated studies during Phase IIB. All subsequent installation sketches sent to Lockheed represented some modification to this design. A detailed description of ejector studies is covered in the Ejector Reversers section of this report.

#### d. Engine Cant and Wing Relationship

During Phase IIB Lockheed moved both the inboard and outboard engines rearward on the wing approximately 5 to 6 feet. Concurrently, they requested that the engine be canted  $4^\circ$  downward. A study was made to determine the most desirable location for canting the engine, and a point just forward of the ejector was chosen. Figure 1-32 (outboard engine) and Figure 1-33 (inboard engine) show the STF219-L-700 engine installed in the Lockheed L-2000-4 wing with a  $4^\circ$  engine cant. This sketch indicated that the rearward relocation of the engine allowed the installation of the engine-ejector to significantly reduce wing blockage of the blow-in doors. During Phase IIA, the forward location of the engine placed the ejector in the thicker part of the wing and resulted in blockage of two blow-in doors.

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e. Reduction of Sonic Boom Overpressure Requirements

The FAA's reduction of the sonic boom overpressure reduced the maximum thrust requirement for the engine. The engine is now sized for cruise thrust levels instead of the greater thrust levels previously required for transonic acceleration. In addition, refinement of the aircraft design with accompanying increases in lift/drag values lowered thrust requirements. As a result the size of the engine was reduced to 650 lbs/sec airflow size.

Figures 1-34 and 1-35 are installation drawings for the new size of engine. Critical areas (i.e. accessory arrangement and ejector O.D.) were reviewed to assure proper nacelle clearance. These drawings reflect the changes in basic engine length derived from the continuing detailed design of the engine. The most significant change was the addition of a diffuser section between the third stage turbine blades and the rear struts. This results in the reduction of the exhaust gas velocity from the turbine to the desired value. This feature is further explained in the section of this report on Turbine Design. A revised weight for the 650 lbs/sec STF219 engine was developed.

f. Miscellaneous Studies

Lockheed expressed concern about the area available for passing secondary airflow between the front mount ring O.D. and the nacelle wall. Information was then supplied to Lockheed describing the weight changes incurred in reducing the front mount ring O.D. by 0.5 and 1.0 inches to increase the secondary air passage. The design of the ring had been optimized in Phase IIA.

While the detail design of the STF219 turbofan continued, a study was made to position the  $P_{t2}$  and  $T_{t2}$  probes, which are necessary for biasing the engine fuel control. The probes were located in the engine front mount ring as shown in Figure 1-36.

At the end of Phase IIA, Lockheed requested a study of flame arrestors in the secondary airstream. Although spring-loaded flapper valves in the secondary air stream to prevent a reverse or forward flow of secondary air were included in the Phase IIA design, it was considered possible that during conditions of very low rearward secondary airflow, a combustible mixture from a leak in the engine compartment might ignite when it reached the hot exhaust section. This could propagate forward through the flapper valve openings into the engine compartment.

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Lockheed provided a sketch of a suggested flame arrestor showing the general construction, frontal area, and desired pressure drop. The resulting study is shown on Figure 1-37. The flame arrestors were located immediately forward of the flapper valves. Their installation requires a mounting structure, attachment flanges, and accessibility for replacement of damaged units. The original study based on the 700-lbs/sec engine showed that the flame arrestors could be installed without difficulty and still provide the required area to maintain a minimum pressure drop.

Continued detail design of the engine and the reduction in engine size from 700 lbs/sec to 650 lbs/sec resulted in slightly less room for the flame arrestors. Similarly, continued ejector studies resulted in a tighter wrap between the engine and the ejector reducing still further the area available for the flame arrestors. A study was made of the entire secondary airflow passage from the engine inlet to the ejector throat because of these changes and the possible increase in secondary airflow to nearly 18 percent of the primary airflow.

Figure 1-38 shows that the minimum area available for secondary airflow occurs at the flame arrestors, flapper valves, and rear mount rings. Figure 1-39 shows more detail of these areas. This minimum area is now smaller than earlier studies showed. With an increase in secondary airflow and a corresponding increase in pressure drop, continued studies will be required in this area to provide a proper installation.

#### g. Turbojet Installations

The reintroduction of the turbojet engine during Phase IIB required updating of the Phase IIA installation. As a result, two preliminary installation drawings for a partial afterburning and a full afterburning turbojet were developed for the initial study (Figures 1-40 and 1-41). Subsequently, as the turbojet design evolved, a series of six installation sketches were prepared each showing the effect of turbine inlet temperature and airflow schedule selection on the size of the engine and ejector. Figure 1-42 presents a typical turbojet engine. The major dimensions and differences between each of the six engines are summarized on page 1-16. All versions of the turbojet are fully augmented and weights for each of the six engines were quoted.

Figure 1-43 shows external dimensions and a proposed accessory arrangement for the high temperature, base flow engine.

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The 4° downward cant of the engine is required for the turbojet engine as well as the turbofan. Various combinations of engine positions relative to the L-2000-4 wing shape were tried. The current design has placed the cant in the turbojet engine slightly aft of the rear mount structure. All subsequent installation studies showed the cant at this location with an octagonal sliding shroud ejector.

The additional length of the turbojet engine plus the rearward movement of the engines relative to the wing makes it possible to install the engine in the L-2000-4 wing without blockage of any of the ejector blow-in doors. Reaching this optimum position requires setting the ejector higher into the wing and increasing the cant angle to approximately 6° (Figure 1-44). As seen in the figure, the engine rear mount ring appears to be in a favorable position for direct mounting to the wing.

An alternate scheme placed the top of the ejector flush with the top of the wing. This allowed full utilization of the blow-in doors and introduced the added feature of reduced cant angle. Unfortunately this arrangement lowered the engine relative to the wing sufficiently to eliminate a practical mounting installation.

Continued detail design of the engine will create further areas requiring coordination and revised installation techniques and procedures.

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TABLE 1-1  
PRINCIPAL INSTALLATION DIMENSIONS  
FOR LOCKHEED CALIFORNIA CORPORATION  
STJ227 TURBOJET ENGINE, 525 LBS./SEC. SIZE  
TRANSLATING SHROUD EJECTOR, 4° CANT ANGLE

Turbine Inlet Temp. (°F)	Airflow Schedule	Inlet Diameter (in.)	Overall Length (in.)	Ejector Equivalent Diameter (in.)	Distance Between Mount Planes (in.)
2000	High Flow	56.30	307.32	78.00	119.15
2000	Base Flow	56.30	305.54	75.00	119.15
2000	Low Flow	56.30	303.56	72.00	119.15
2300	High Flow	56.30	299.54	78.00	121.15
2300	Base Flow	56.30	296.24	75.00	121.15
2300	Low Flow	56.30	296.26	70.00	121.15

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**B. ENGINE CYCLE STUDIES**

**1. INTRODUCTION**

Two engines were studied during Phase II B, the STF219, a non-mixed flow duct-heating turbofan, and the STJ227, an afterburning turbojet. The design parameters for these engines are presented in Table 1-2.

**TABLE 1-2**

**Design Parameters of Afterburning Turbojet (STJ227)  
and Duct-Heating Turbofan (STF219) Engines**

	<u>STJ227</u>		<u>STF219</u>	
Turbine Inlet Temperature (°F)				
Take-Off and Acceleration	2000	2300	2000	2300
Cruise	1900	2200	1900	2200
Corrected Airflow (lb/sec)				
Nominal	525	525	650	650
Cruise (Mach = 2.7)	372	372	308	308
Over-All Pressure Ratio	9.3	9.3	11.9	11.9
Fan Pressure Ratio			2.7	2.7
Bypass Ratio			1.3	1.3
Maximum Engine Diameter (inches)	75	75	81	81
Engine Weight Including Ejector (lb)	10470	10455	9560	9560

During Phase II A, the airframe manufacturers were supplied with sufficient information to permit them to make preliminary engine-aircraft performance estimates. At this time, both airframe manufacturers selected the STF219 engine. Boeing initially expressed an interest in the STJ227 engine with 25 percent augmentation, but this engine exceeded the limits for stable afterburner combustion above Mach 2, being specifically designed for augmentation during transonic operation from Mach 1.2 to Mach 1.8. Above Mach 2, the combustion chamber Mach number increased rapidly above the combustion stability limit and the combustion chamber inlet temperature dropped into the marginal auto ignition range. However, during Phase II B, the sonic overpressure

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Limits were increased from 2.0 to 2.5 paf for transonic acceleration and from 1.5 to 1.7 paf for supersonic cruise. This shift resulted in the engine being sized for supersonic cruise rather than for transonic acceleration and made the STJ227 turbojet engine with full augmentation an attractive powerplant. Consequently, both engines were restudied.

The performance of the uninstalled engines at several critical flight conditions is shown in Table 1-3. The thrust specific fuel consumption over a wide range of power settings is shown in Figures 1-45 through 1-47. The effect of reoptimizing the STF219 engine is shown in Table 1-4 and Figures 1-48 through 1-50. Similar data for the STJ227 engine is presented in Table 1-5 and Figures 1-51 through 1-53. Detailed design information is presented in Section 2.

TABLE 1-3

Performance of Uninstalled  
STJ227 and STF219 Engines

	Afterburning Turbojet STJ227		Duct Heating Turbofan STF219	
Take-Off Maximum Thrust (lb)	57000	59800	52500	57000
Nominal Airflow Size (lb/sec)	525	525	650	650
Cruise Turbine Inlet Temperature (°F)	1900	2200	1900	2200
<u>Transonic Acceleration, Mach 1.2, at 45000 Ft</u>				
Thrust (lb)	20600	21400	18400	19800
Specific Fuel Consumption (lb/hr/lb)	1.91	1.83	1.88	1.84
<u>Supersonic Cruise Mach 2.7 at 65000 Ft, Thrust = 9800 lb</u>				
Specific Fuel Consumption (lb/hr/lb)	1.44	1.45	1.54	1.48
<u>Subsonic Part Throttle, Mach 0.9 at 36150 Ft, Thrust = 7500 lb</u>				
Specific Fuel Consumption (lb/hr/lb)	1.05	1.06	0.88	0.91
<u>Subsonic Part Throttle, Mach 0.6 at 15000 Ft, Thrust = 6500 lb</u>				
Specific Fuel Consumption (lb/hr/lb)	1.25	1.25	0.95	0.96

Source: 1-10

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TABLE 1-4

Performance of STEAR Engine at Completion of Phase IIA  
and at Completion of Phase IIB

	Basic Engine		Initial Engine	
	Phase IIA	Phase IIB	Phase IIA	Phase IIB
Supersonic Cruise Turbine inlet Temperature (°F)	2200	2200	1900	1900
Take-Off Maximum Thrust (lb)	56800	57000	52300	52500
Airflow Size (lb/sec)	650	650	650	650
Maximum Diameter (in)	79.7	81	79.7	81
Engine Weight Including Ejector (lb)	9150	9560	9200	9560
<u>Transonic Acceleration, Mach 1.2 at 45000 Ft</u>				
Thrust (lb)	19700	19800	18300	18400
Specific Fuel Consumption (lb/hr/lb)	1.84	1.84	1.88	1.88
<u>Supersonic Cruise, Mach 2.7 at 65000 Ft, Thrust = 9800 lb</u>				
Specific Fuel Consumption (lb/hr/lb)	1.49	1.48	1.55	1.54
<u>Subsonic Part Throttle, Mach 0.9 at 36150 Ft, Thrust = 7500 lb</u>				
Specific Fuel Consumption (lb/hr/lb)	0.92	0.91	0.89	0.88
<u>Subsonic Part Throttle, Mach 0.6 at 15000 Ft, Thrust = 6500 lb</u>				
Specific Fuel Consumption (lb/hr/lb)	0.97	0.96	0.96	0.95

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TABLE 1-5

Performance of STJ227 Engine at Completion of Phase IIA  
and at Completion of Phase IIB

	Basic Engine Phase IIA	Phase IIB	Initial Engine Phase IIB
Supersonic Cruise Turbine Inlet Temperature ( $^{\circ}\text{F}$ )	2200	2200	1900
Take-Off Maximum Thrust (lb)	60600	59800	57000
Airflow Size (lb/sec)	525	525	525
Maximum Diameter (in)	78	75	75
Engine Weight Including Ejector (lb)	9850	10455	10470
<u>Transonic Acceleration, Mach 1.2 at 45000 Ft</u>			
Thrust (lb)	21400	21400	20600
Specific Fuel Consumption (lb/hr/lb)	1.86	1.83	1.91
<u>Supersonic Cruise, Mach 2.7 at 65000 Ft, Thrust 9800 lb</u>			
Specific Fuel Consumption (lb/hr/lb)	1.50	1.45	1.44
<u>Subsonic Part Throttle, Mach 0.9 at 36150 Ft, Thrust = 7500 lb</u>			
Specific Fuel Consumption (lb/hr/lb)	1.10	1.06	1.05
<u>Subsonic Part Throttle, Mach 0.6 at 15000 Ft, Thrust = 6500 lb</u>			
Specific Fuel Consumption (lb/hr/lb)	1.33	1.25	1.25

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Particular emphasis during Phase IIB was placed on the augmentation systems, and IBM performance decks were provided to the airframe manufacturers to permit airframe-powerplant optimization with the fully augmented turbojet engine. In addition, a study of inflight thrust measurement techniques based on the STF219 engine was conducted.

## 2. AUGMENTATION SYSTEM STUDIES

The relaxation of the sonic boom overpressure limit permitted a re-optimization of the engine cycles, and, in particular, the augmentation systems. For the STF219 engine, the effect of reducing the amount of augmentation and the magnitude of the thrust discontinuity between un-augmented and minimum augmented operation were studied. For the STF227 engine, the effects of the turbine inlet temperature and engine air flow on augmentor performance were studied.

### a. STF219

Since the engines are now sized for supersonic cruise rather than for transonic acceleration, the fuel consumption during climb can be reduced by reducing the duct heater augmentation thrust and temperature. A reduction in the maximum augmentation thrust of 1 to 5 percent corresponds to an augmentation temperature reduction of 100 to 300°F and increases the aircraft range by 20 to 35 miles.

The Phase IIA evaluation indicated that the evaluating team considered the thrust discontinuity between non-augmented and minimum augmented thrust to be too great. Consequently, a study was conducted to determine the g loading applied to a passenger when the duct heaters on all four engines are lit simultaneously with several augmentation ratios. The results are plotted in Figure 1-54. The upper curve shows the g loading for the performance data presented during Phase IIA for which the fuel-air ratio for lighting was 0.01. For these conditions, the g loading would be only 0.05. However, studies conducted during Phase IIB have indicated that the minimum duct heating fuel-air ratio could be reduced to 0.006, thereby reducing the g loading to 0.04. Stable combustion has been demonstrated at much lower fuel-air ratios, and, with additional development, it would be possible to decrease the minimum fuel-air ratio to 0.003, corresponding to a maximum g loading of 0.01. The problem has been discussed with airframe manufacturers, however, and it was learned that passengers on present commercial jet aircraft experience a g loading of about 0.1, and, in some cases, they experience g loadings as high as 0.33. It appears, therefore, that the g loading produced during the SST

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flight will be significantly lower than that of current jet aircraft and, therefore, that there is little reason to develop a duct heater capable of ignition with a fuel-air ratio below 0.008.

b. STJ227

(1) Effect of Turbine Inlet Temperature

The engines were designed for operation with a turbine inlet temperature of 2200°F. However, initially the engines will probably be operated with a turbine inlet temperature of 1900°F, and, therefore, the effect of operating at the lower turbine inlet temperature must be considered. If the augmentation ratio or thrust increase is to be maintained at the lower turbine inlet temperature, the weight and maximum engine diameter are significantly increased. If the cycle pressure ratio is to remain unchanged, the turbine expansion ratio must be increased to provide the required compressor power. However, increasing the expansion ratio increases the turbine exit Mach number and the afterburner combustion chamber Mach number. Consequently, the engine diameter must be increased to provide acceptable afterburner inlet conditions. The effect of turbine inlet temperature on the maximum engine diameter is shown in Figure 1-55.

With a turbine inlet temperature of 1900°F, unaugmented thrust at cruise is marginal and the capability for augmentation during cruise at nonstandard temperatures or at off-design altitudes is desirable. The IBM decks supplied to the airframe manufacturers, therefore, provided performance, weight, and installation data for engines with augmentation limited to take-off and transonic operation; limited to take-off, acceleration, and supersonic cruise; or unlimited and used throughout the mission. Optimization studies conducted by the airframe manufacturers and based on these data resulted in the selection of the fully augmented version of the STJ227 engine.

(2) Effect of Supersonic Cruise Airflow

The engine corrected airflow at cruise affects the augmentor design in a manner similar to that of the turbine inlet temperature. In order to increase the airflow, the power output of the turbine must be increased, and, therefore, the expansion ratio must be increased. Consequently, the maximum engine diameter must be increased to satisfy the augmentor design criteria.

Continued reduction of the supersonic cruise airflow eventually results in the turbine power requirement being established at acceleration

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conditions rather than at cruise conditions. This effect is shown in Figure 1-56.

### 3. INFLIGHT PERFORMANCE MEASUREMENT SYSTEM

Inflight thrust measurement in supersonic vehicles is considerably more complicated than in subsonic vehicles. In subsonic vehicles, engine pressure ratio, speed, and exhaust temperature have been used in conjunction with a series of charts to indicate the performance and condition of the engine. These parameters provide an accurate method of determining the performance of engines with fixed nozzle geometry. For the more sophisticated engines required for the supersonic transport, however, these parameters are not adequate since the inlet, the exhaust nozzle, and the interference effects between the propulsion system and the airframe significantly affect the actual force applied to the vehicle. Consequently, thrust measurement will require the use of additional parameters in conjunction with a compact, lightweight (about 30 pounds) computer such as those currently being used in missile guidance systems. The parameters required are listed in Table 1-6 together with the measurement accuracy.

Table 1-6

#### Inflight Performance Measurement Parameters

<u>Parameter</u>	<u>Accuracy (Percent)</u>
$P_{am}$ , free stream static pressure	$\pm 1.25$
$P_{t3}$ , fan discharge total pressure	$\pm 1.25$
$P_{s3}$ , fan discharge static pressure	$\pm 1.25$
$T_{t3}$ , fan discharge total temperature	$\pm 1.0$
$(P_{t3} - P_{s3}) / P_{s3}^*$	$\pm 4.0$
$P_{t7}$ , gas generator total exhaust pressure	$\pm 1.0$
$A_{jduct}$ , duct exhaust nozzle throat area **	$\pm 3.0$
$W_f$ , engine fuel flow	$\pm 0.5$

#### Notes:

- \* This parameter may be obtained from engine control where it is used for duct exhaust nozzle positioning.
- \*\* Direct measurement of  $A_{jduct}$  may not be required.

The thrust and thrust specific fuel consumption obtained from these measurements could be in error by as much as the arithmetical sum of the individual errors, but it is unlikely that all errors would be

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at a maximum in the same direction. It is most probable that the error will be that predicted by the root-mean-square method of error summation. The probable error predicted by this method is shown in Table 1-7.

Table 1-7

Probable Error in Inflight Performance  
Measurement

<u>Flight Condition</u>	<u>Power Setting</u>	<u>Probable Error (Percent)</u>	
		<u>Total Thrust</u>	<u>TSFC</u>
Sea-Level Take-Off	Maximum Duct Heating	3.0	3.1
Transonic Acceleration (Mach 1.2)	Maximum Duct Heating	4.2	4.3
Supersonic Cruise (Mach 2.7)	Partial Duct Heating		12.3
Subsonic Cruise (Mach 0.9)	Part Throttle		5.5

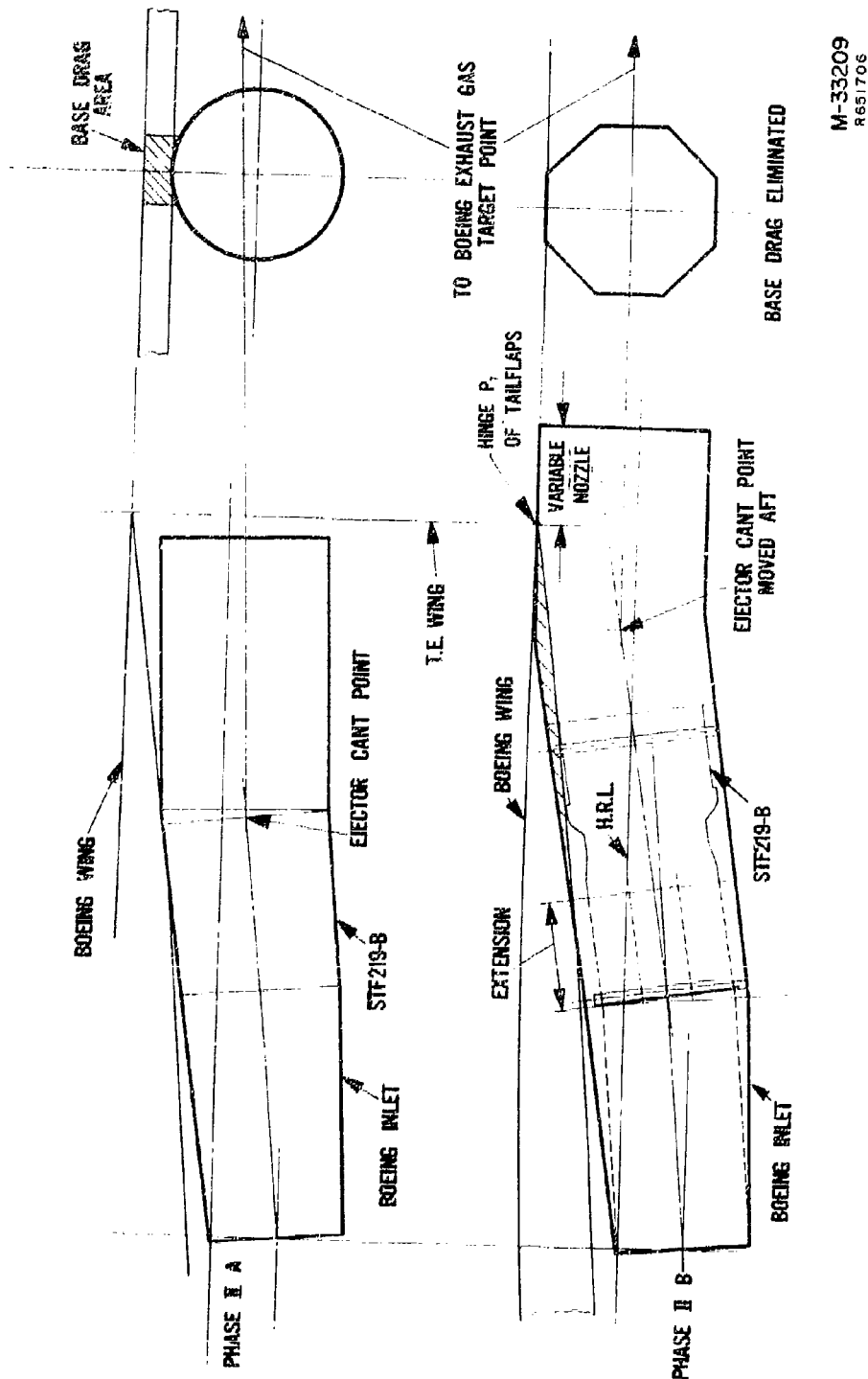
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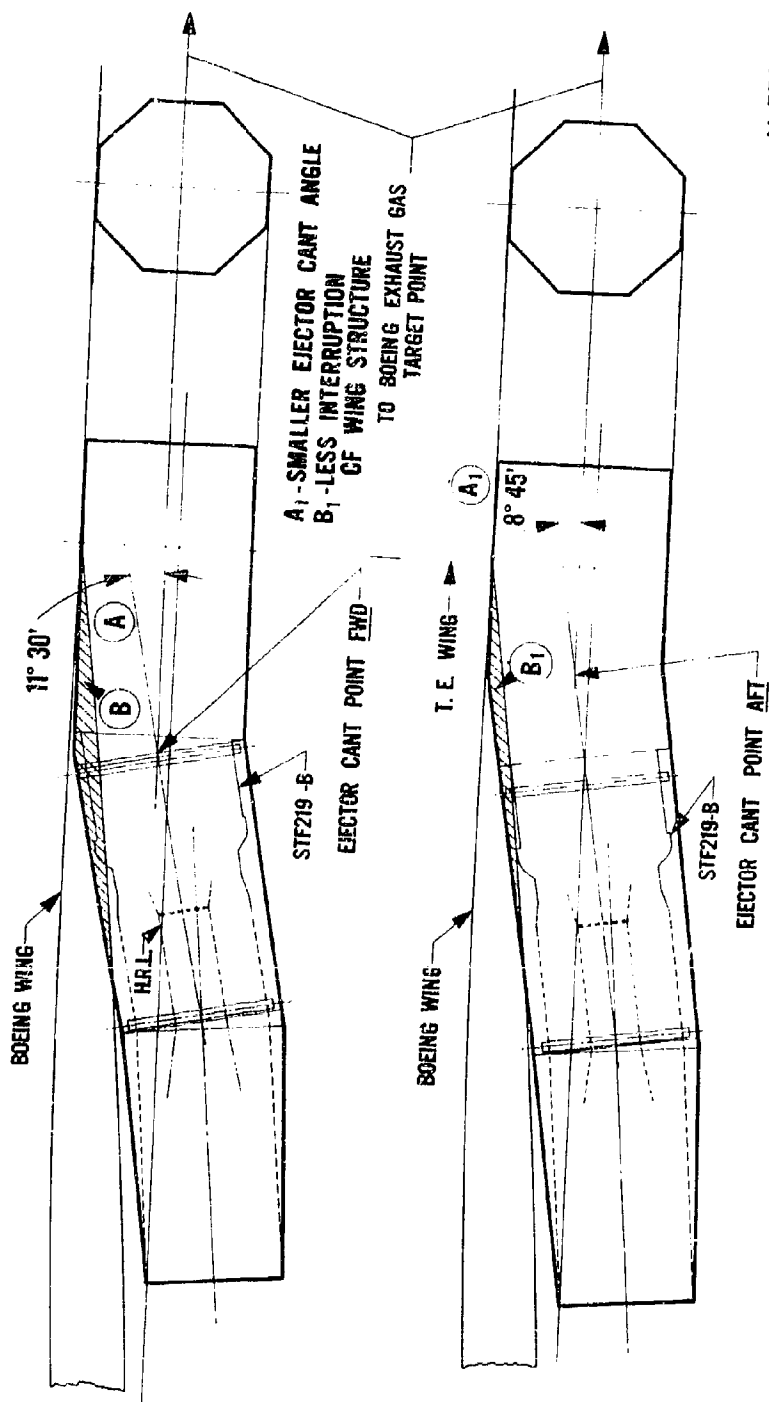
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Figure 1-1

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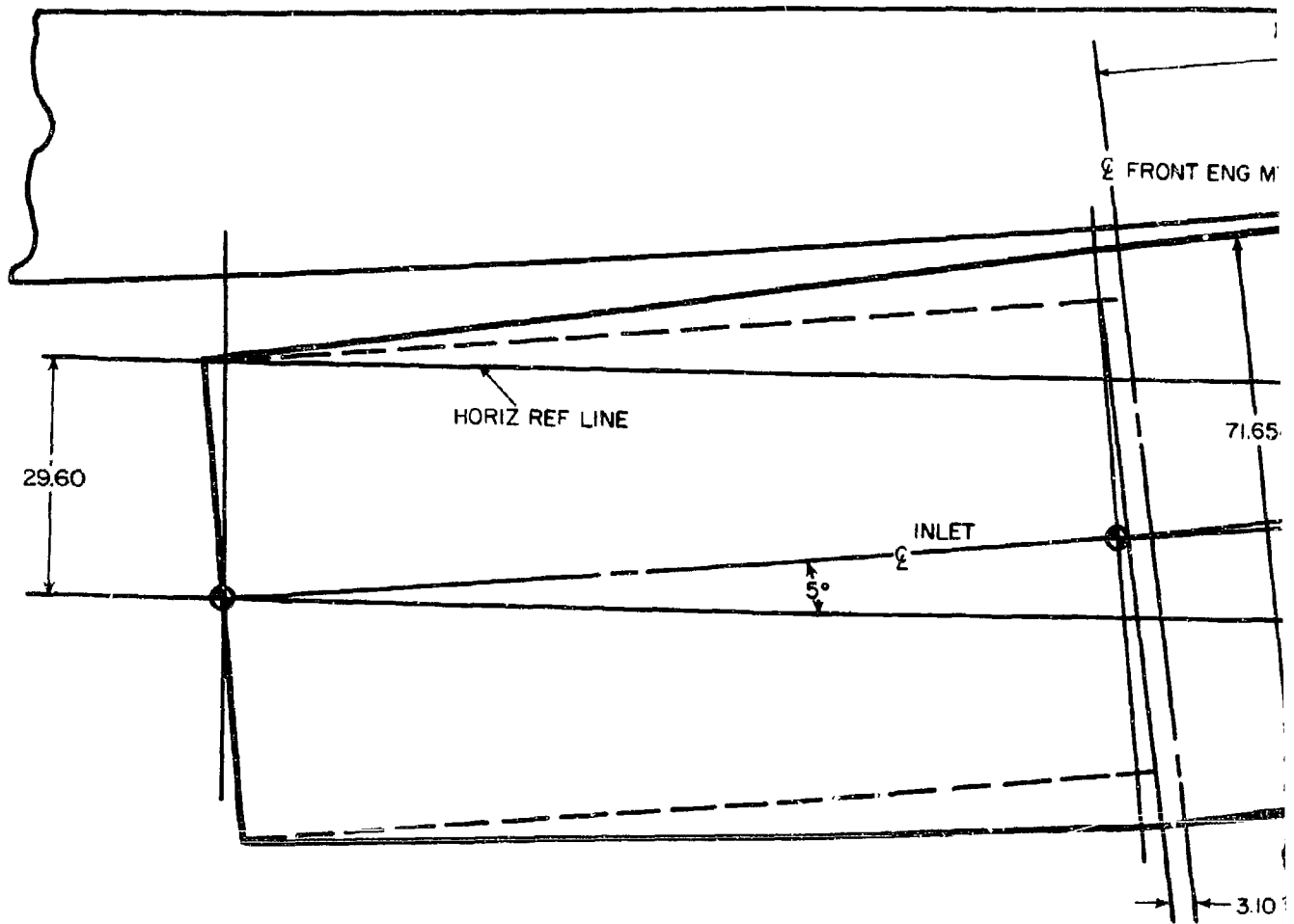


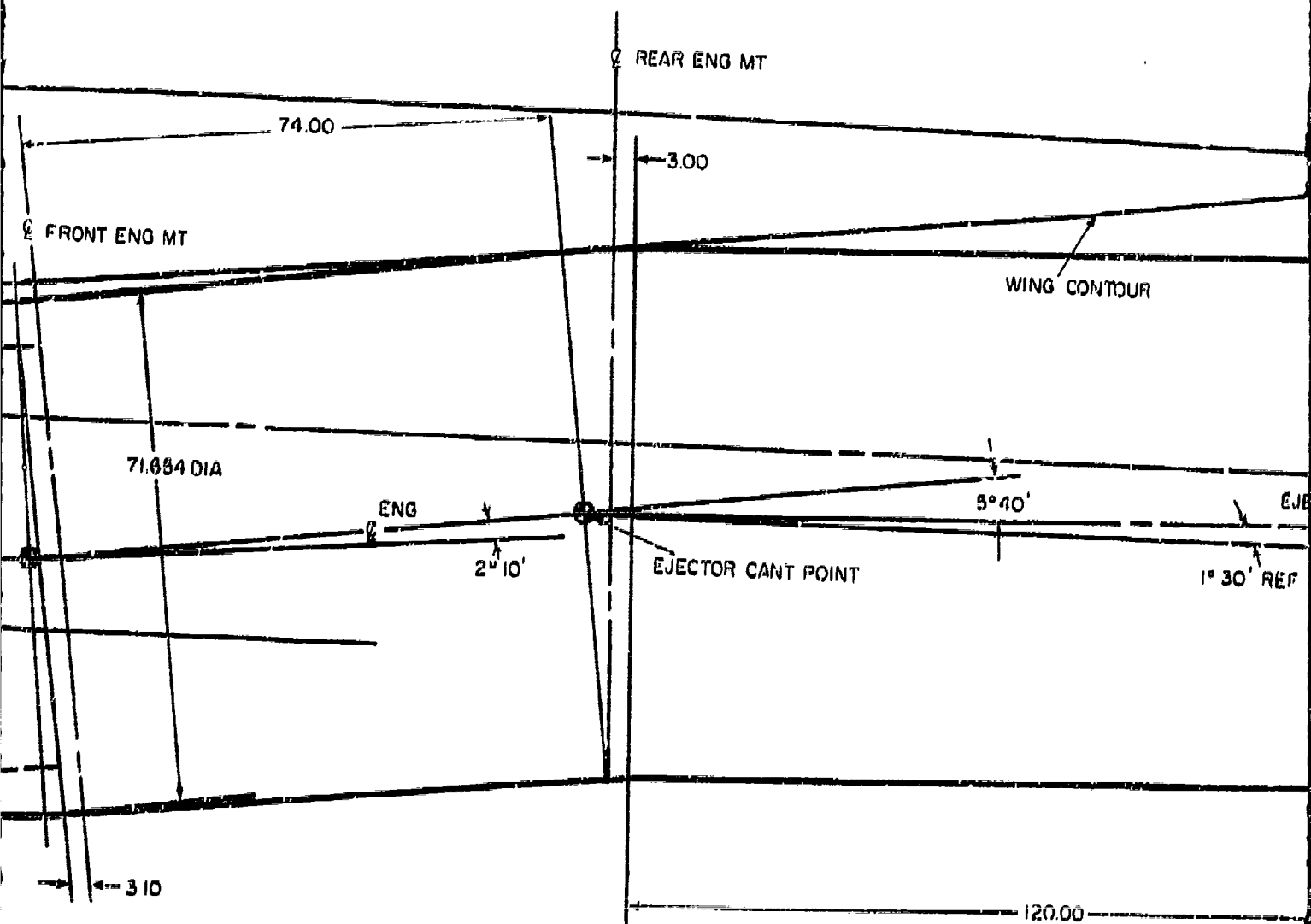
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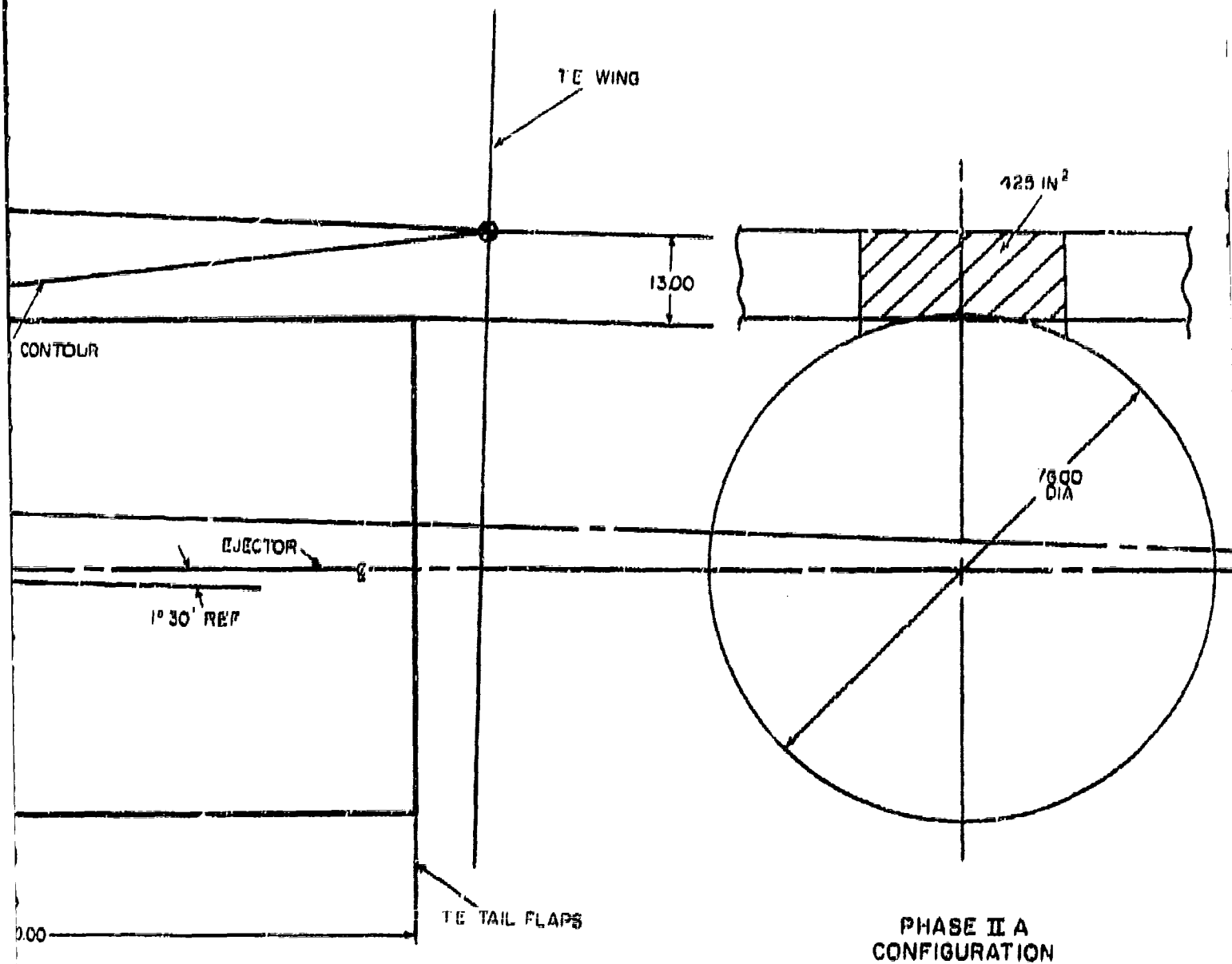
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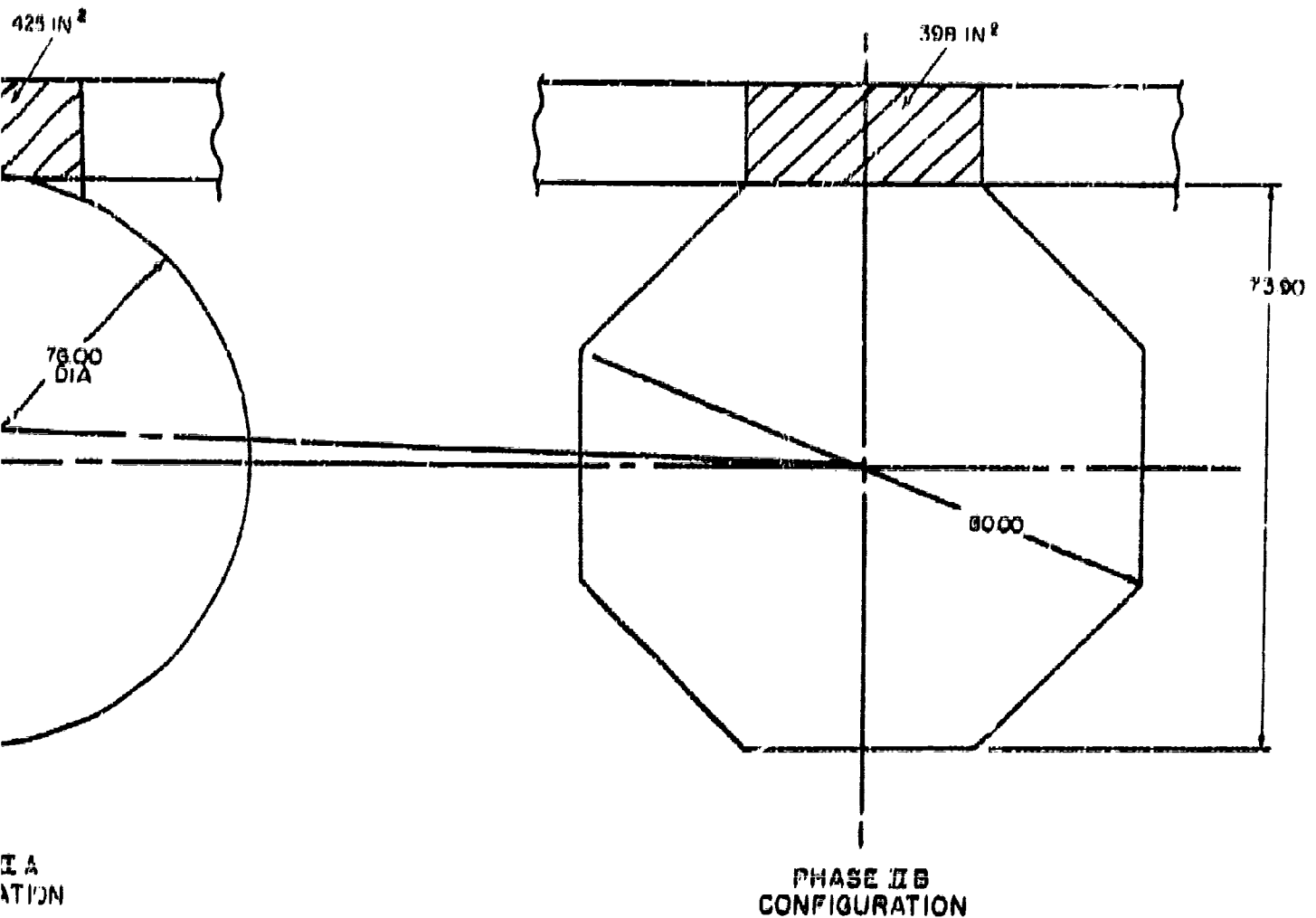






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PHASE I A  
CONFIGURATION

PHASE II B  
CONFIGURATION

ENGINE + EJECTOR TANGENT TO WING  
EJECTOR CANTED AT REAR ENG. MOUNT PLANE  
NOTE: ENGINE MOUNT PLANES ARE  
NONPARALLEL & THE TAIL FLAPS  
ARE FWD. OF THE WING  
BASE DRAG AREA 425 in<sup>2</sup> (EQUIV. ROUND)  
BASE DRAG AREA 398 in<sup>2</sup> (OCT.)

ST7219B 600 LBS. /SEC. TURBOFAN

Figure 1-3

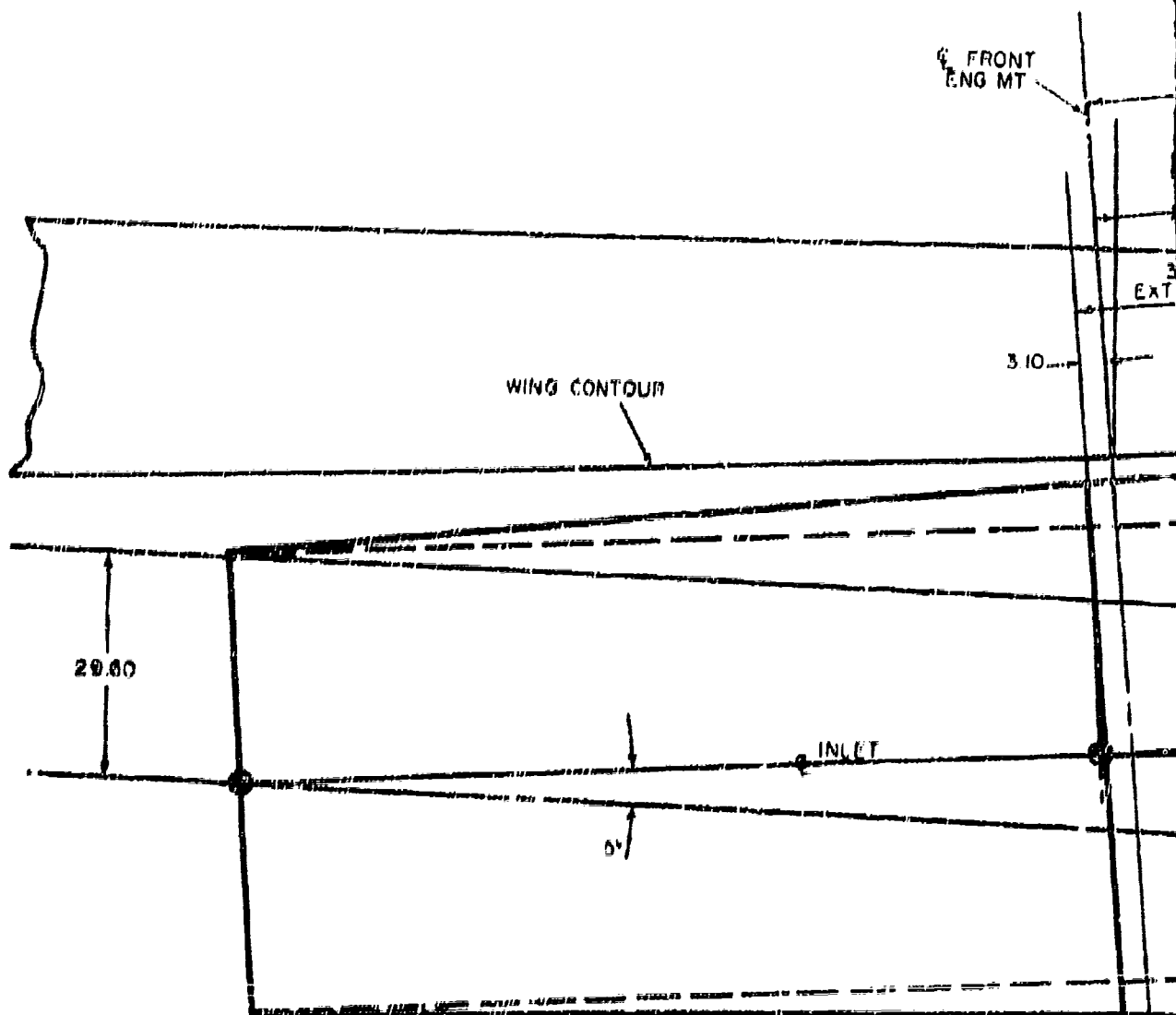
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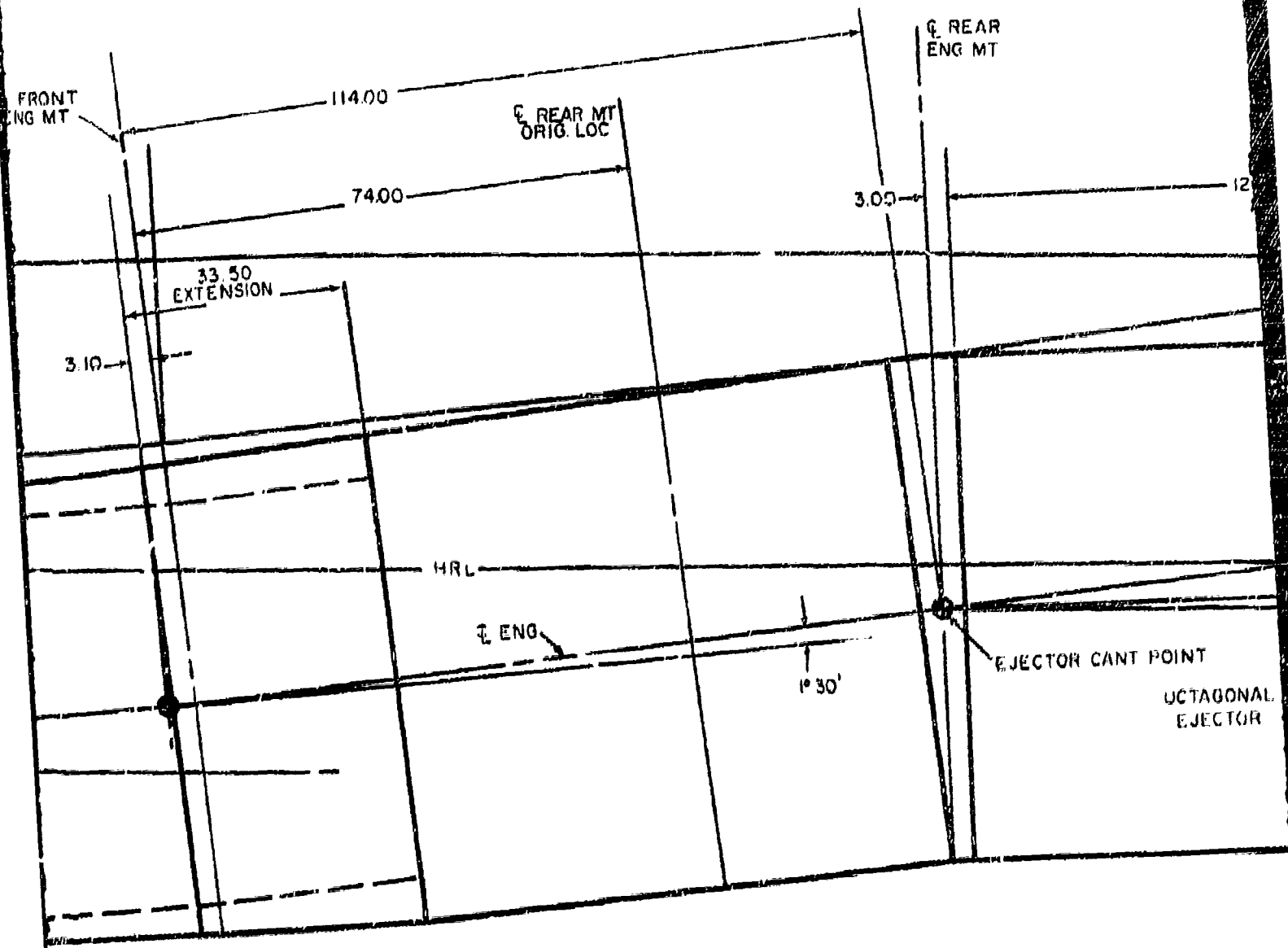
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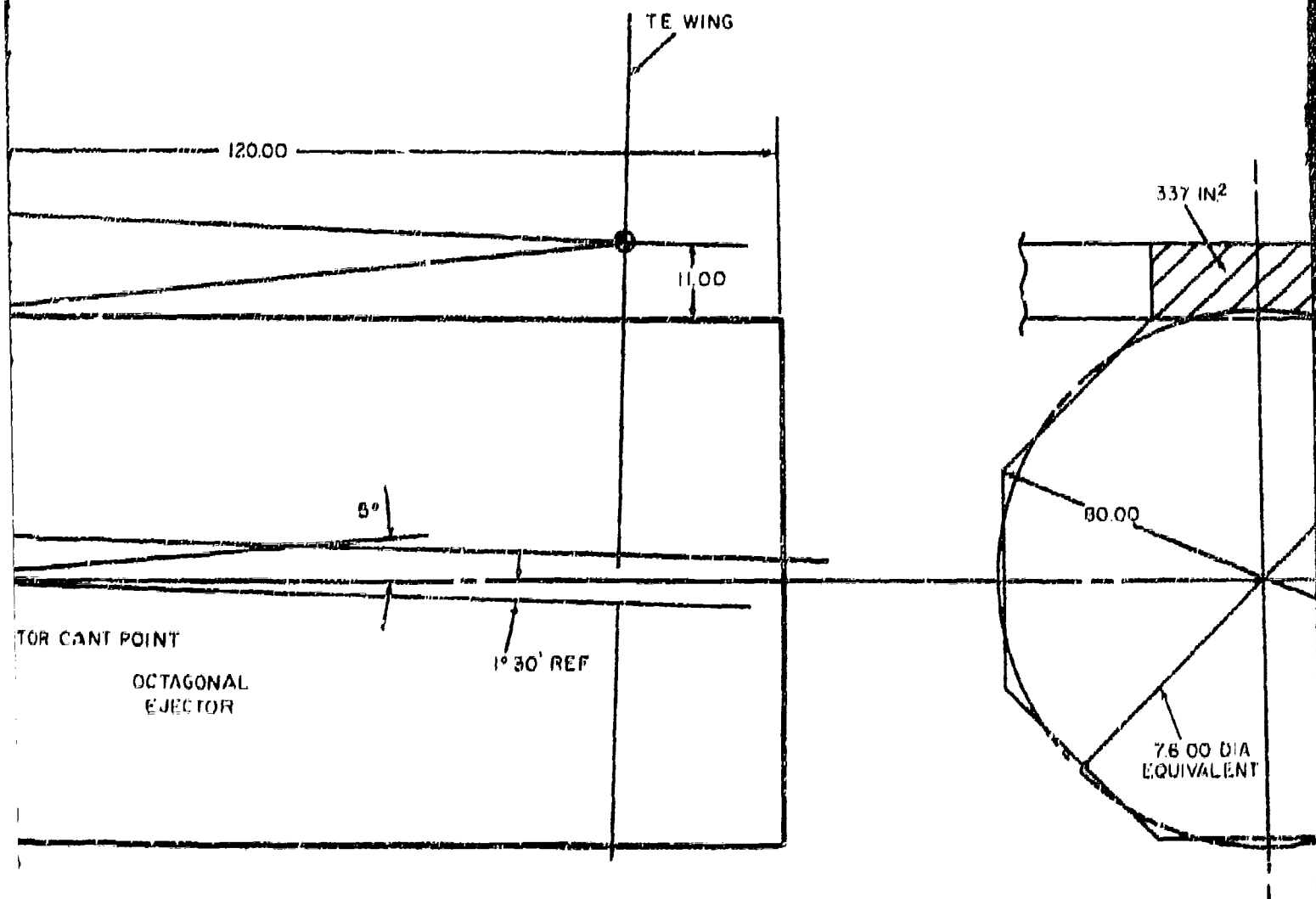
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ARRANGEMENT SHOWING RE  
 MOVED AFT 40.00  
 ENGINE & EJECTOR ARE TA  
 EJECTOR CANTED AT REA  
 PLANE  
 ENGINE, MOUNT PLANES /  
 BASE DRAG AREA 337 in<sup>2</sup>

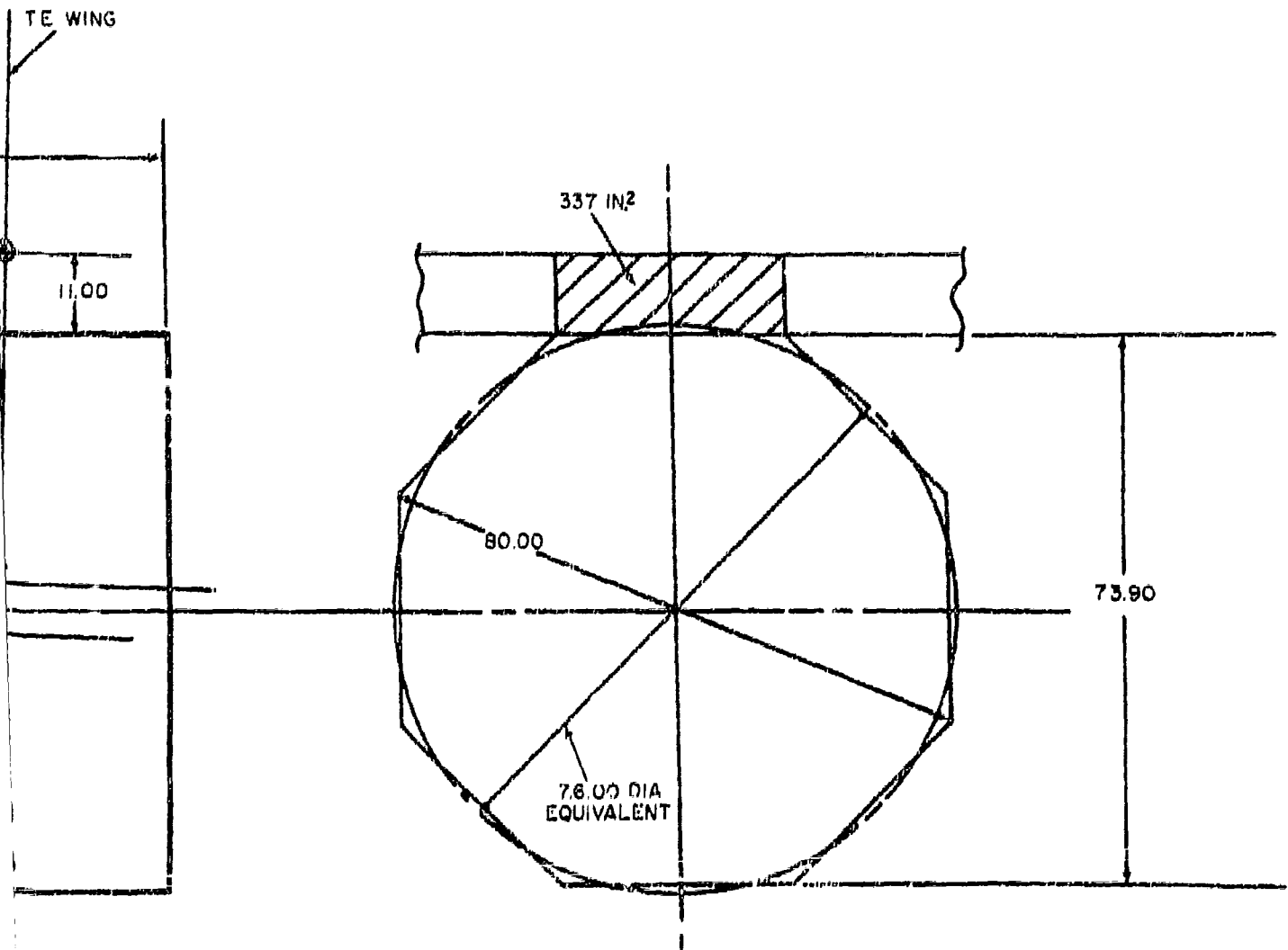
STF219B 600 LBS. /SEC

Figure 1

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ARRANGEMENT SHOWING REAR ENG. MOUNT  
MOVED AFT 40.00  
ENGINE & EJECTOR ARE TANGENT TO WING  
EJECTOR CANTED AT REAR ENG. MOUNT  
PLANE  
ENGINE, MOUNT PLANES ARE NONPARALLEL  
BASE DRAG AREA 337 in<sup>2</sup>

STF219B 600 LBS./SEC. TURBOFAN

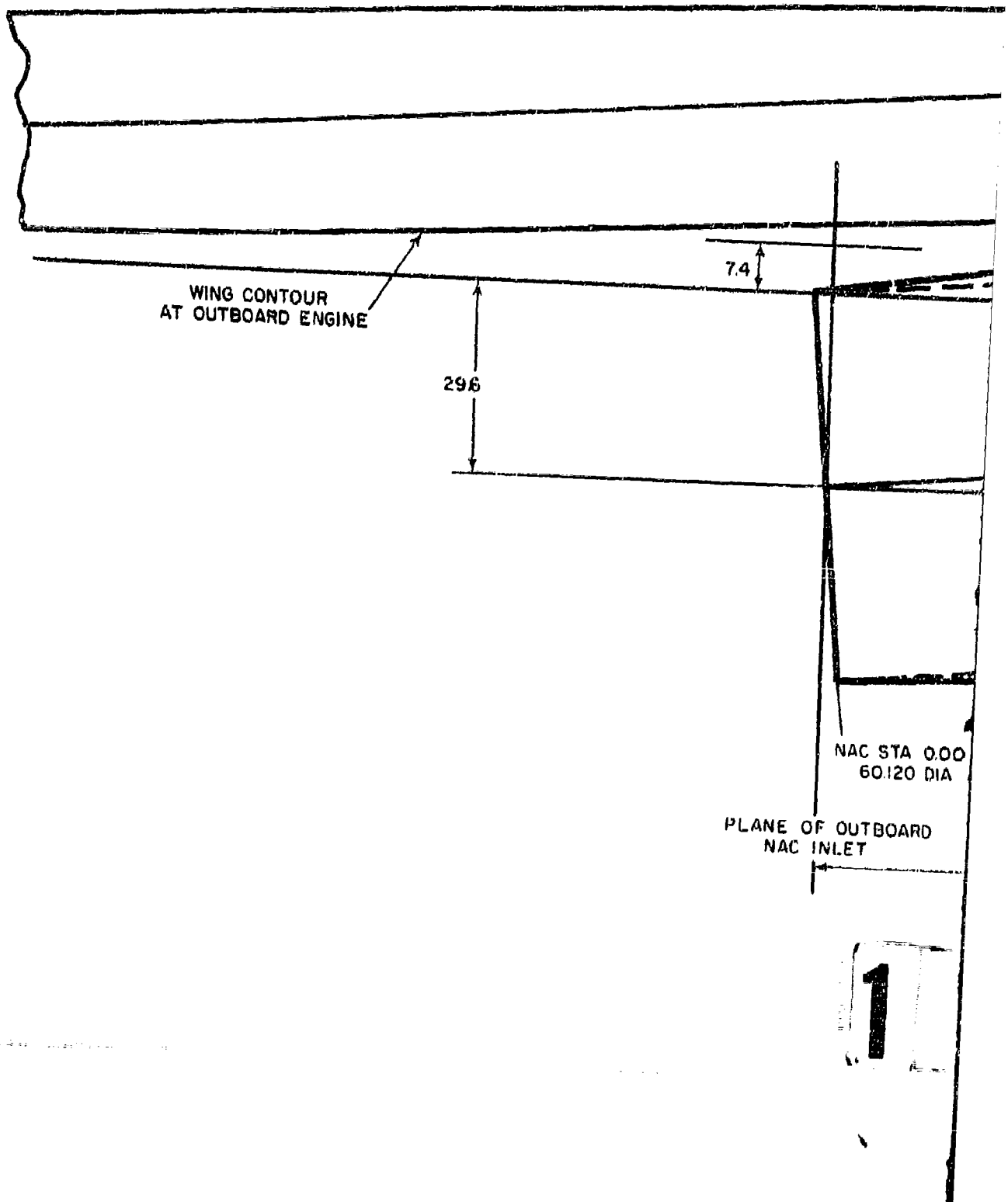
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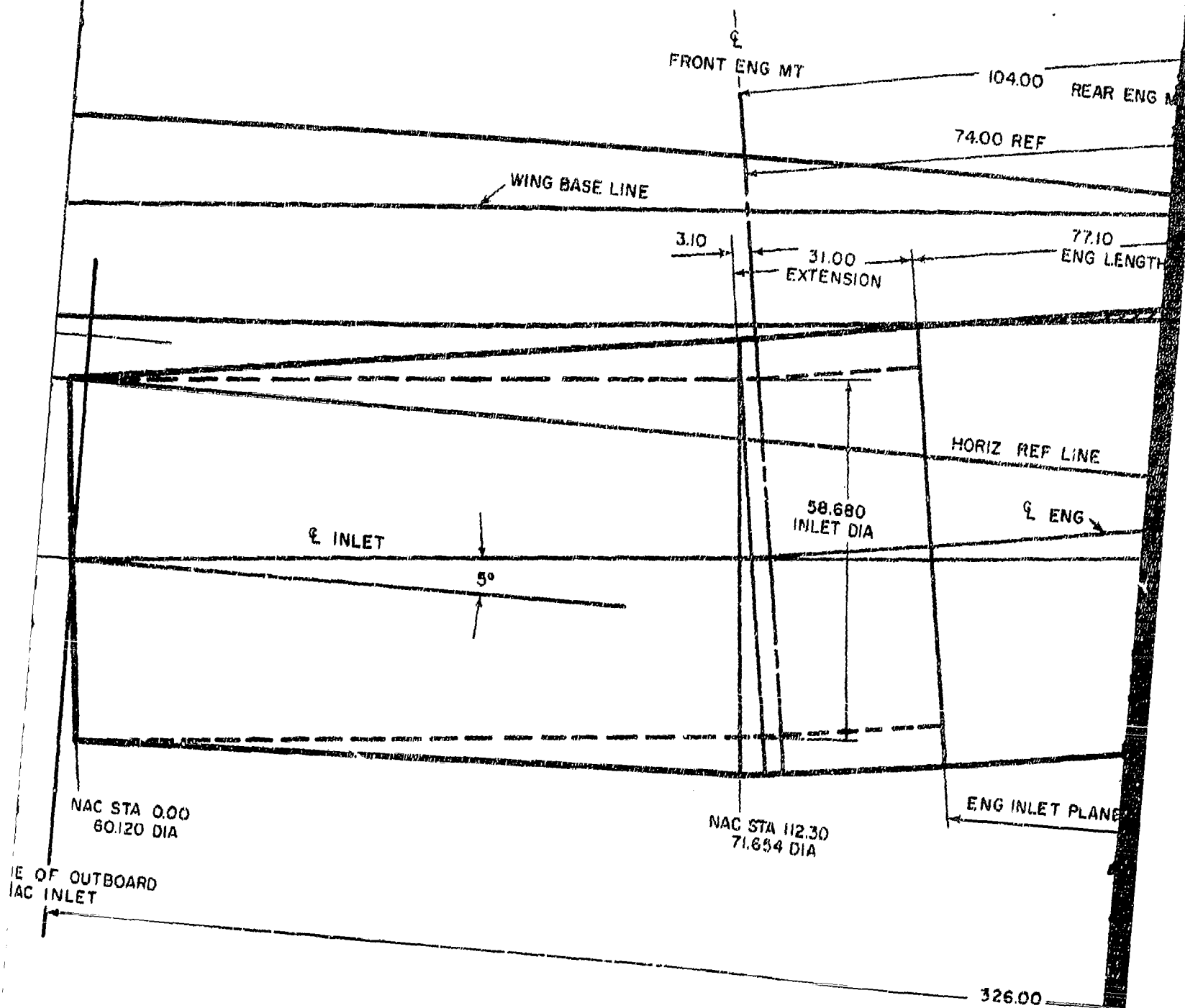
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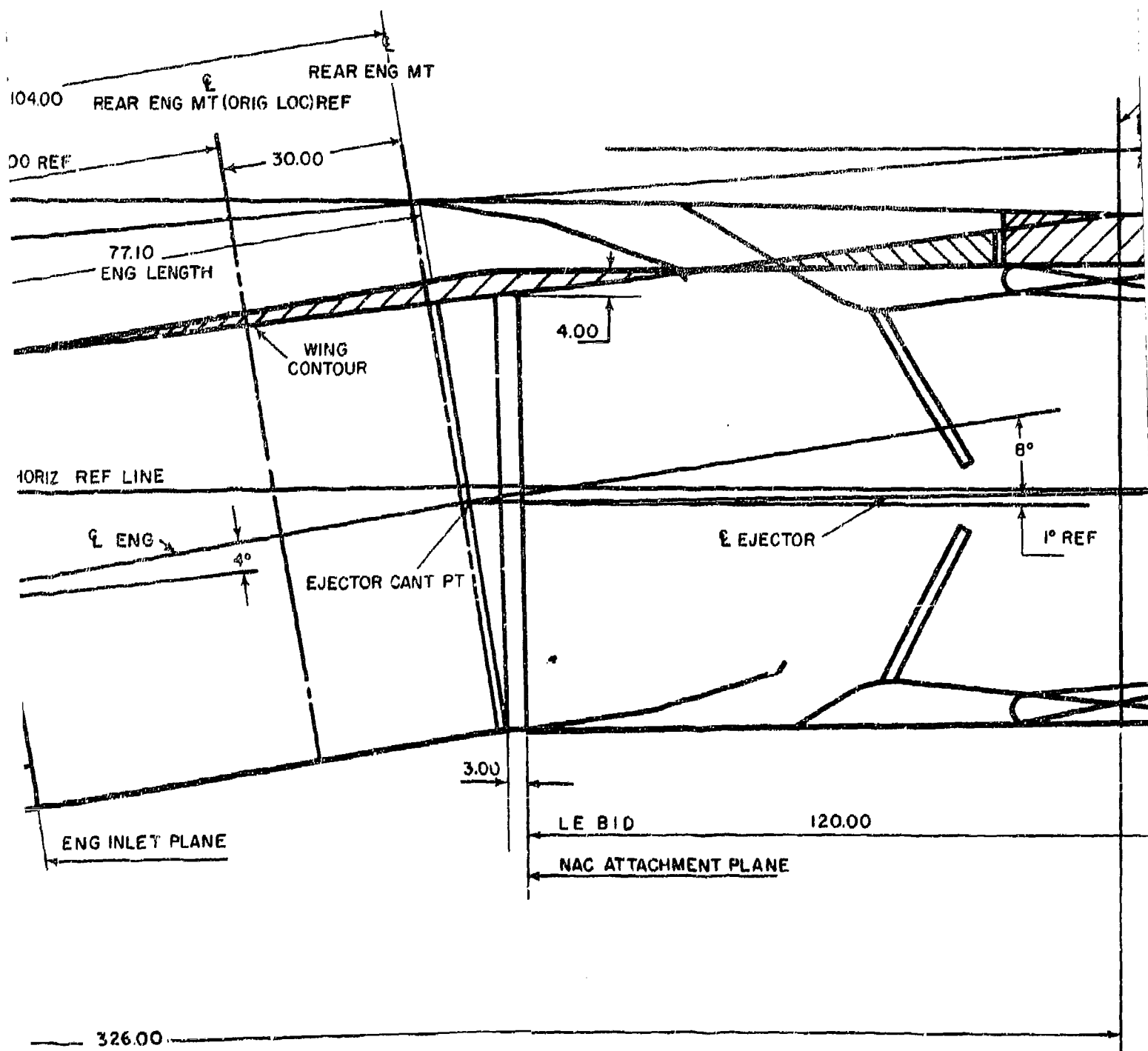
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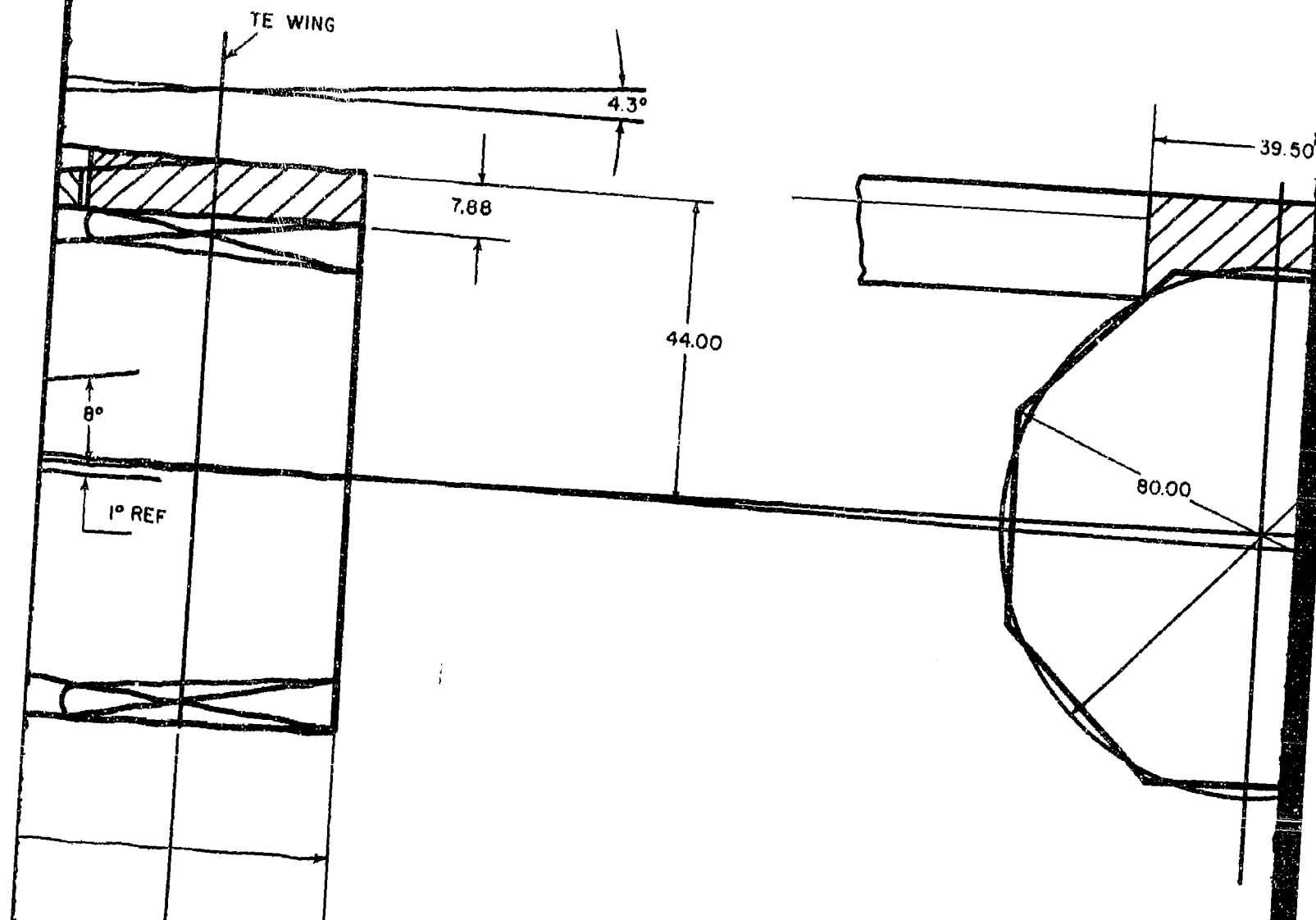








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EJECTOR CANTED AT REAR MOUNT PLANE  
ARRANGEMENT MOVING REAR ENG. MOUNT  
30.00 AFT. & INSERTING ENG. 4.00 INTO  
WING

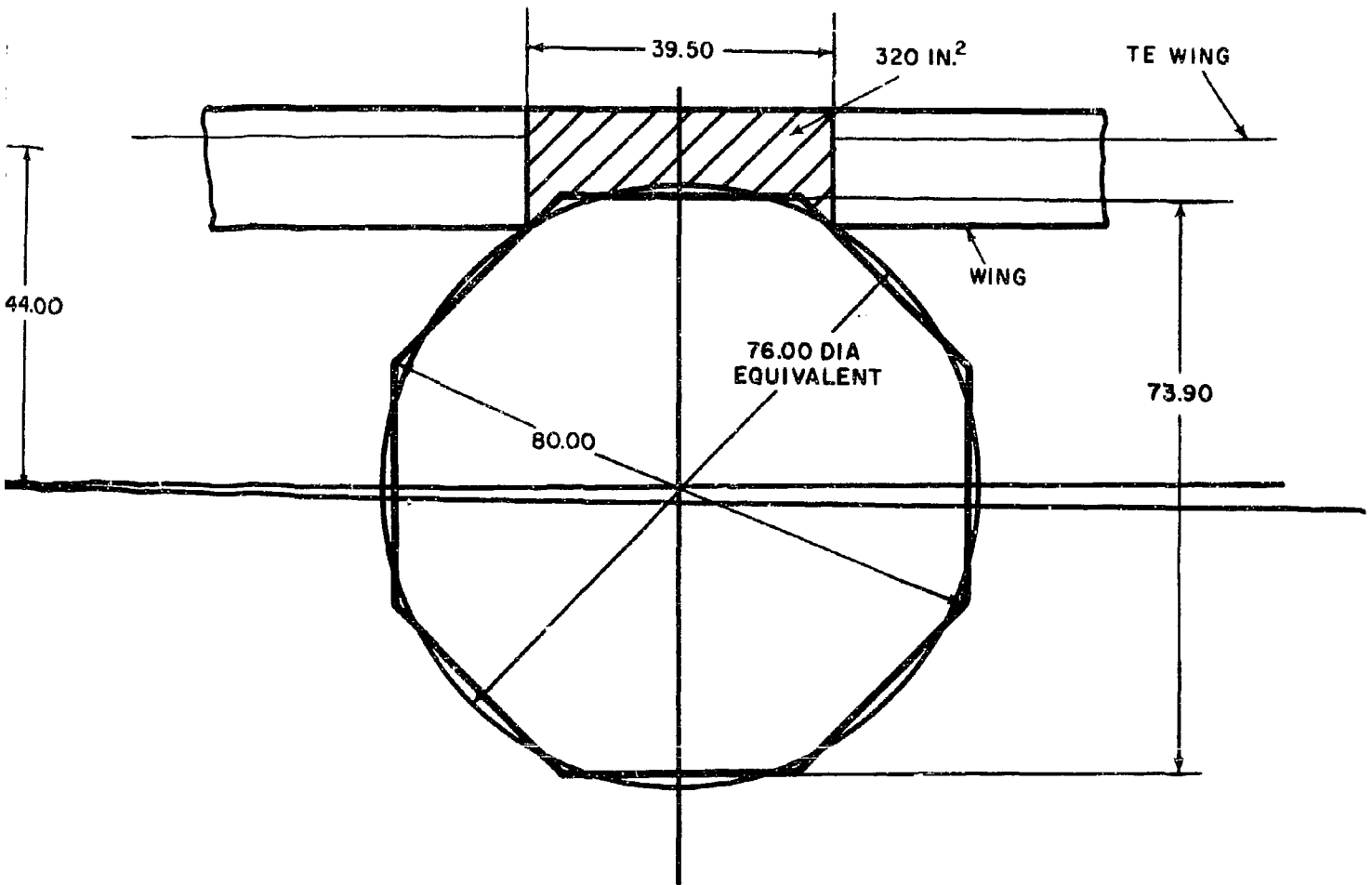
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EJECTOR CANTED AT REAR MOUNT PLANE  
ARRANGEMENT MOVING REAR ENG. MOUNT  
30.00 AFT. & INSERTING ENG. 4.00 INTO  
WING

STF219B 600 LBS./SEC. TURBOFAN

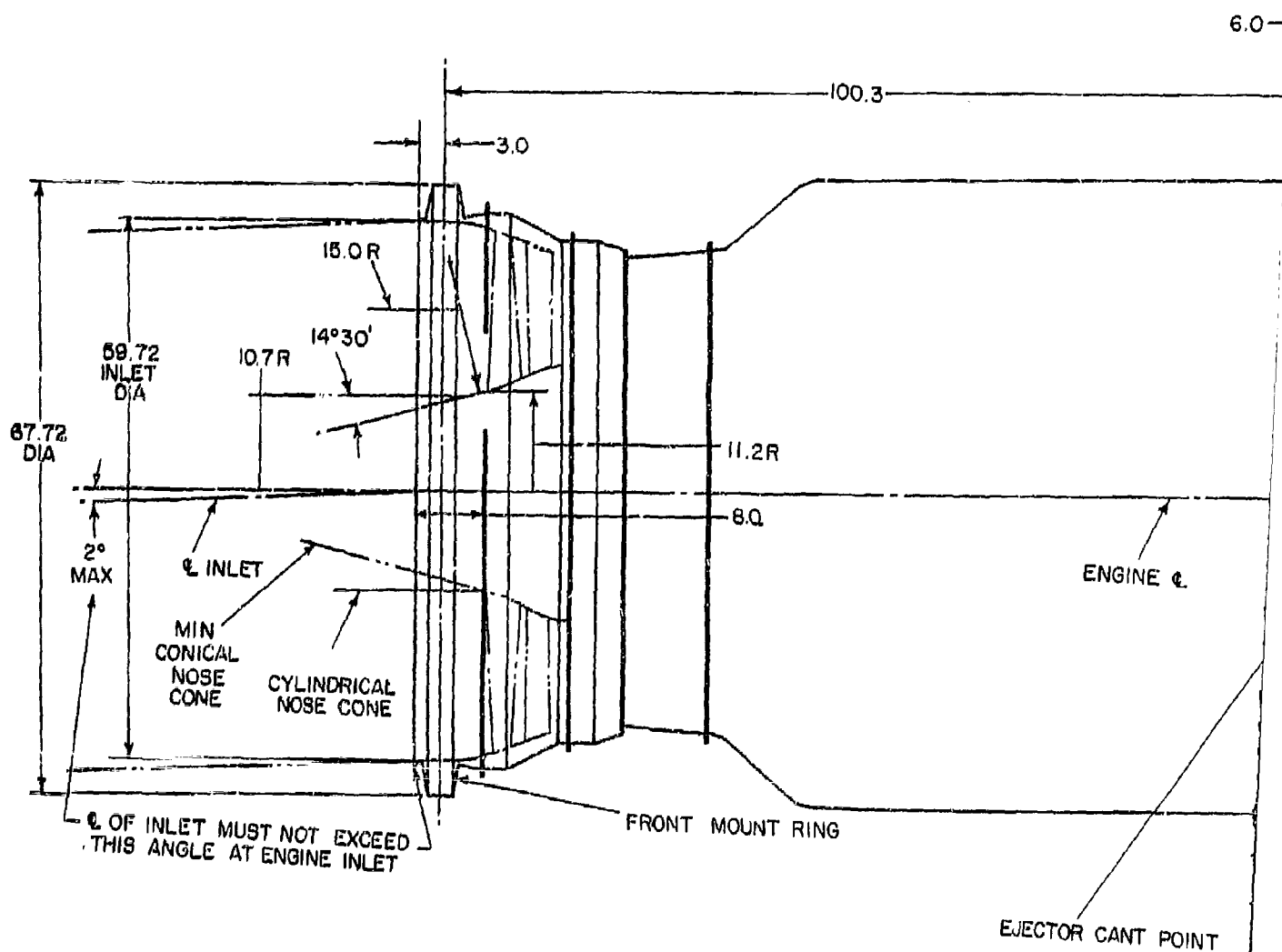
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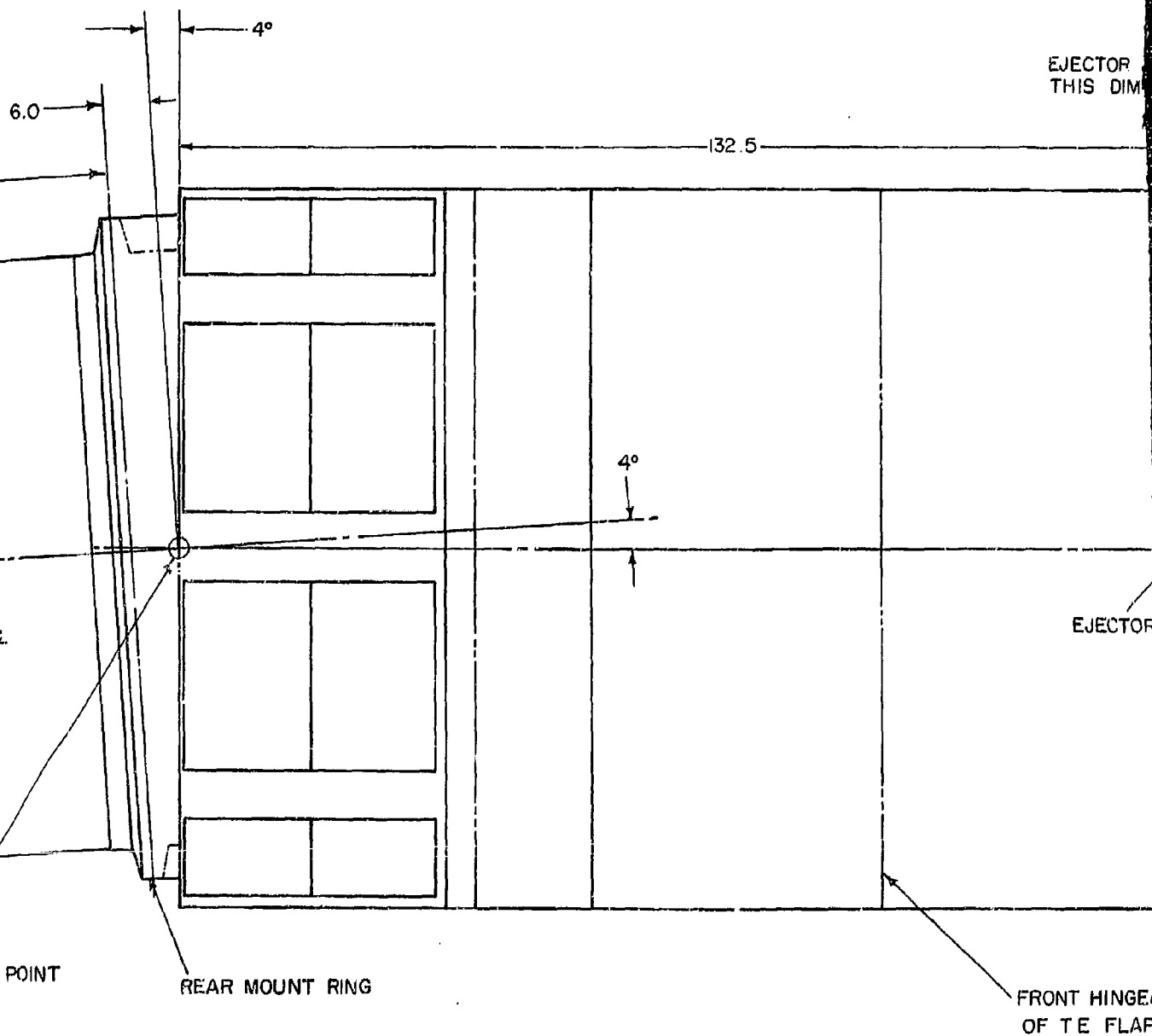
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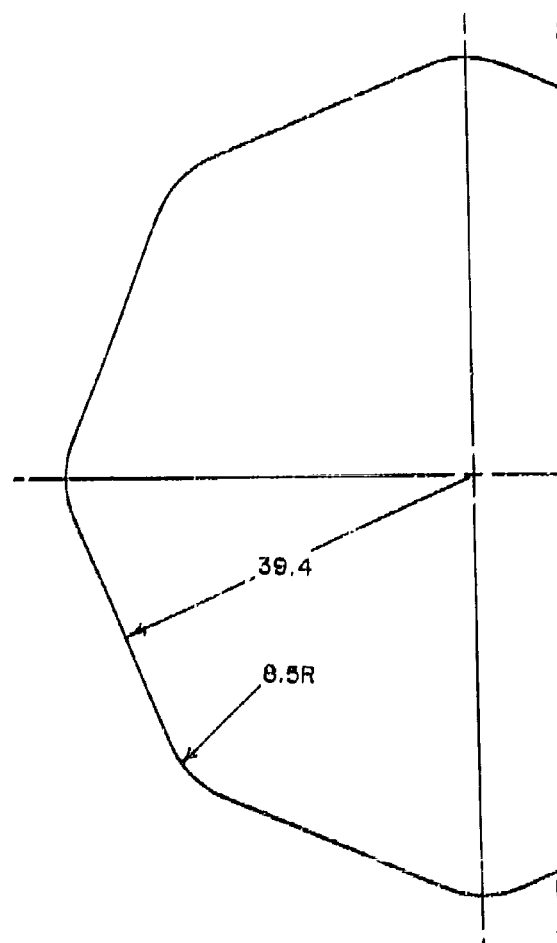
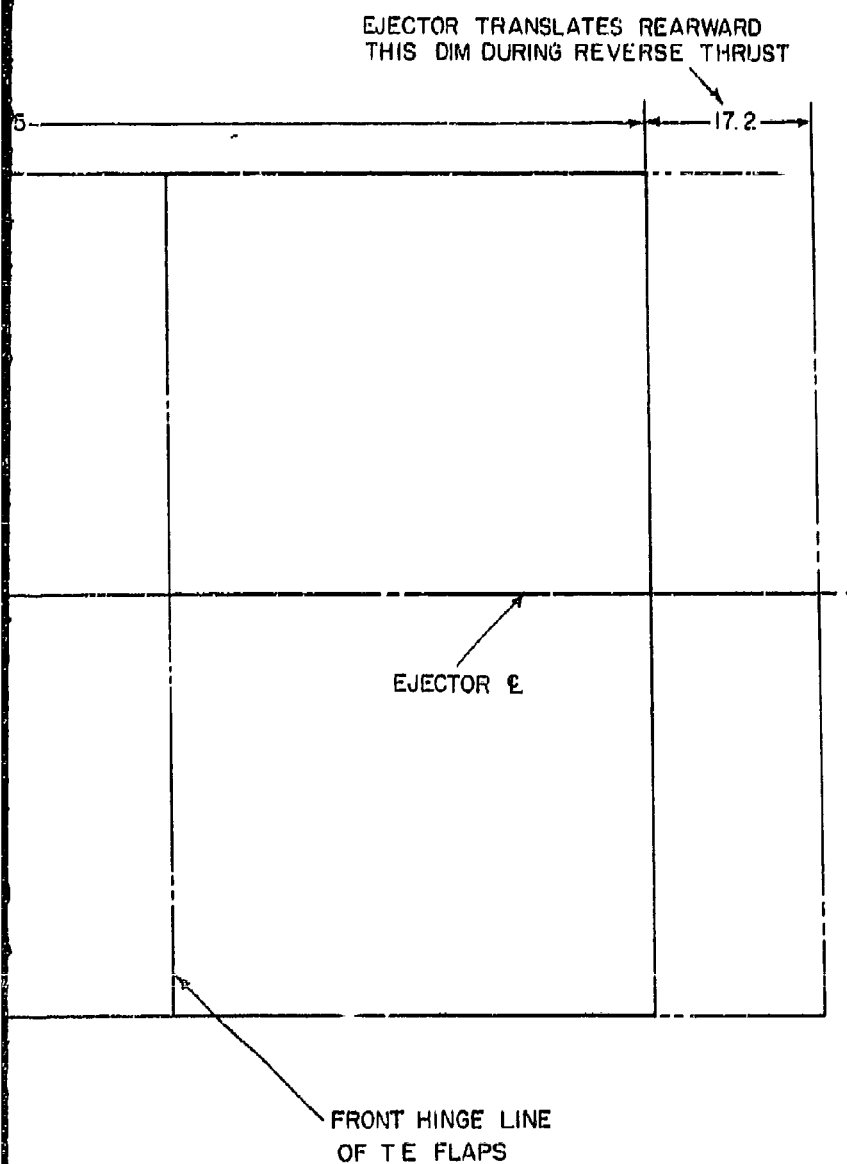
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EJECTOR CANTED AT REAR  
MOUNT PLANE

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Figure 1-6

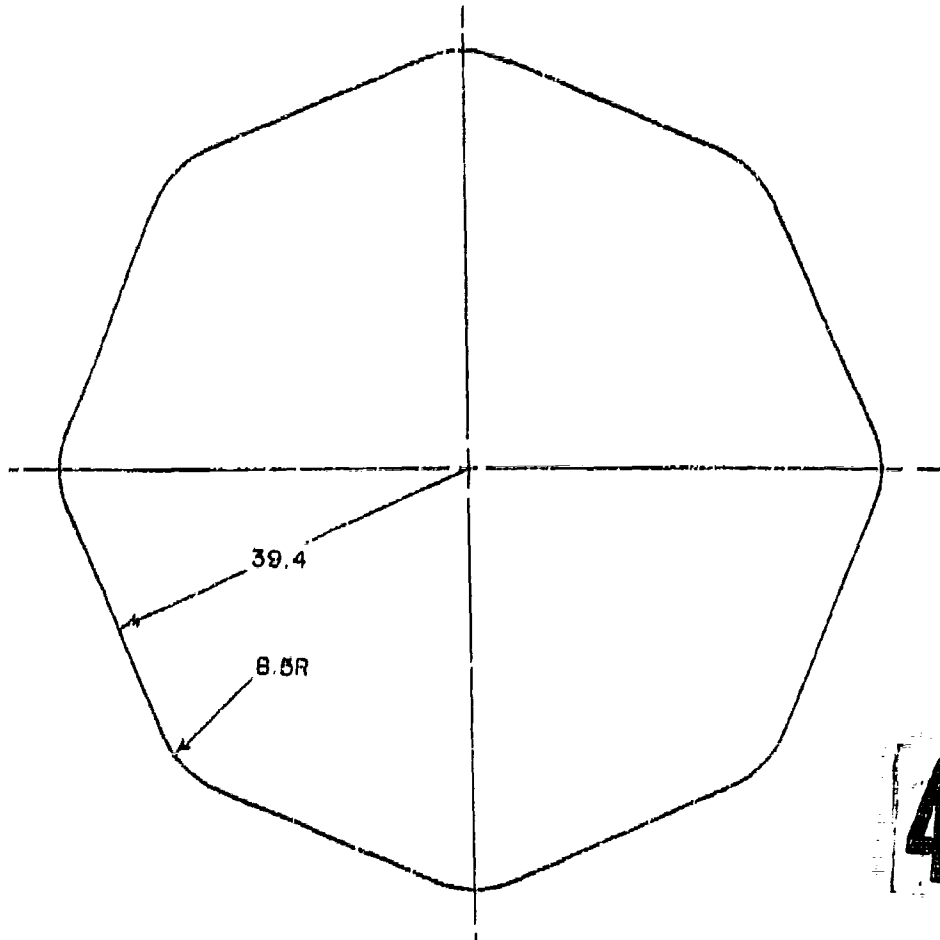
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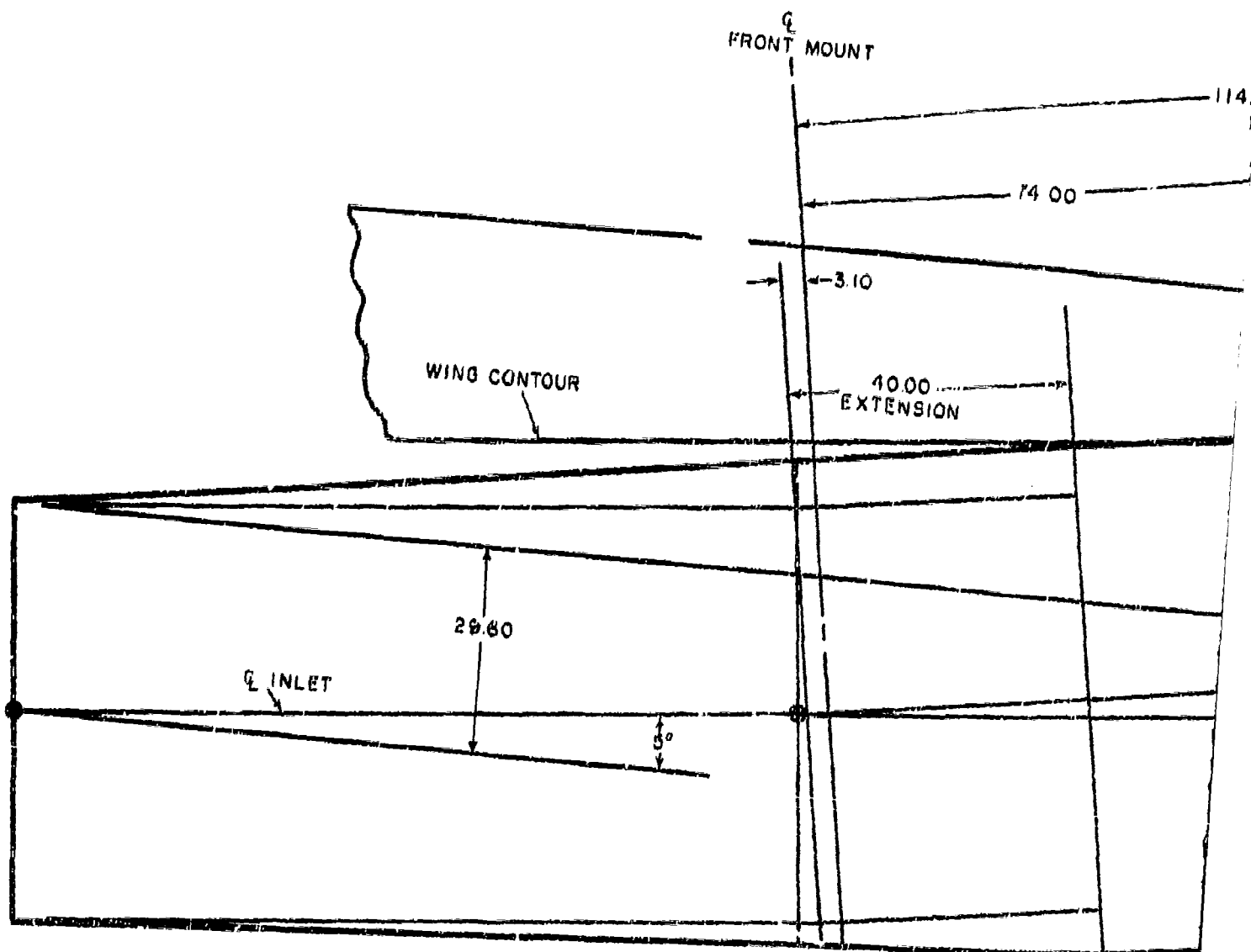
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Figure 1-6

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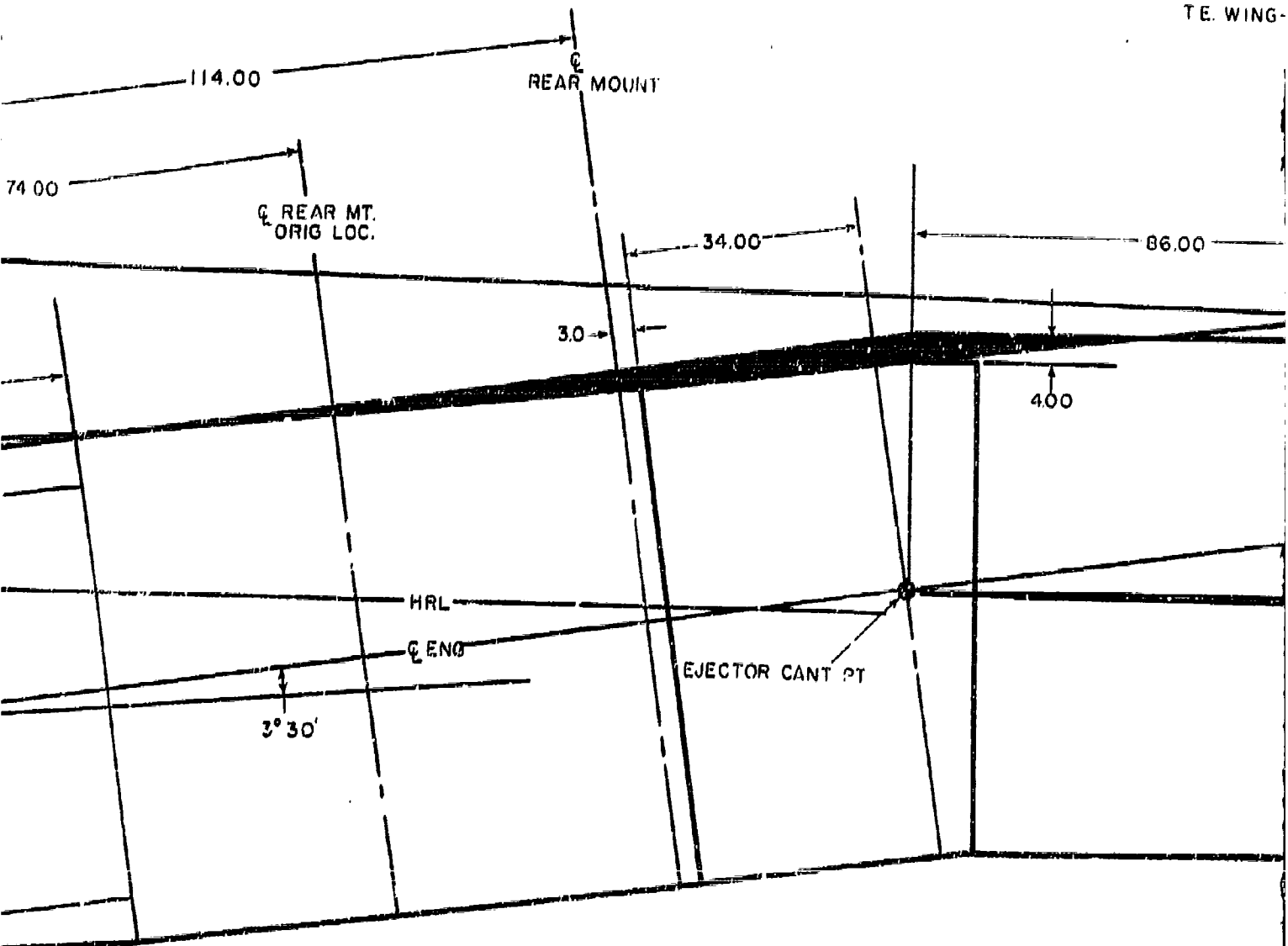
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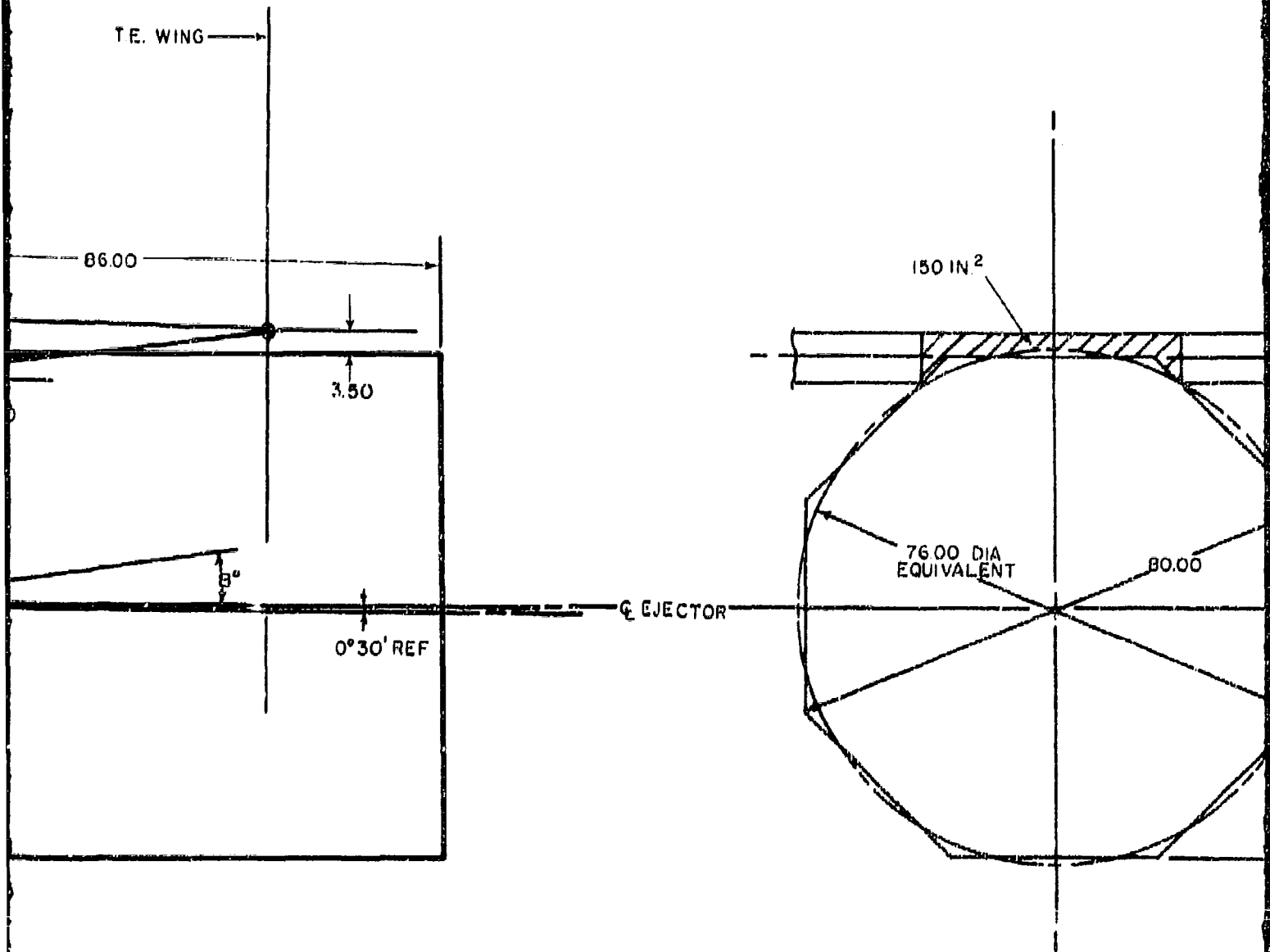




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ARRANGEMENT MOVING REAR ENG. MT  
40.00 & INSERTING ENG. 4.00 INTO WIN  
EJECTOR CANTED AT NOZZLE PLANE  
ENG. MOUNT PLANES ARE PARALLEL  
BASE DRAG AREA = 150 in<sup>2</sup>

STF219B 600 LBS. /SEC. TURBOFAN

Figure 1-7

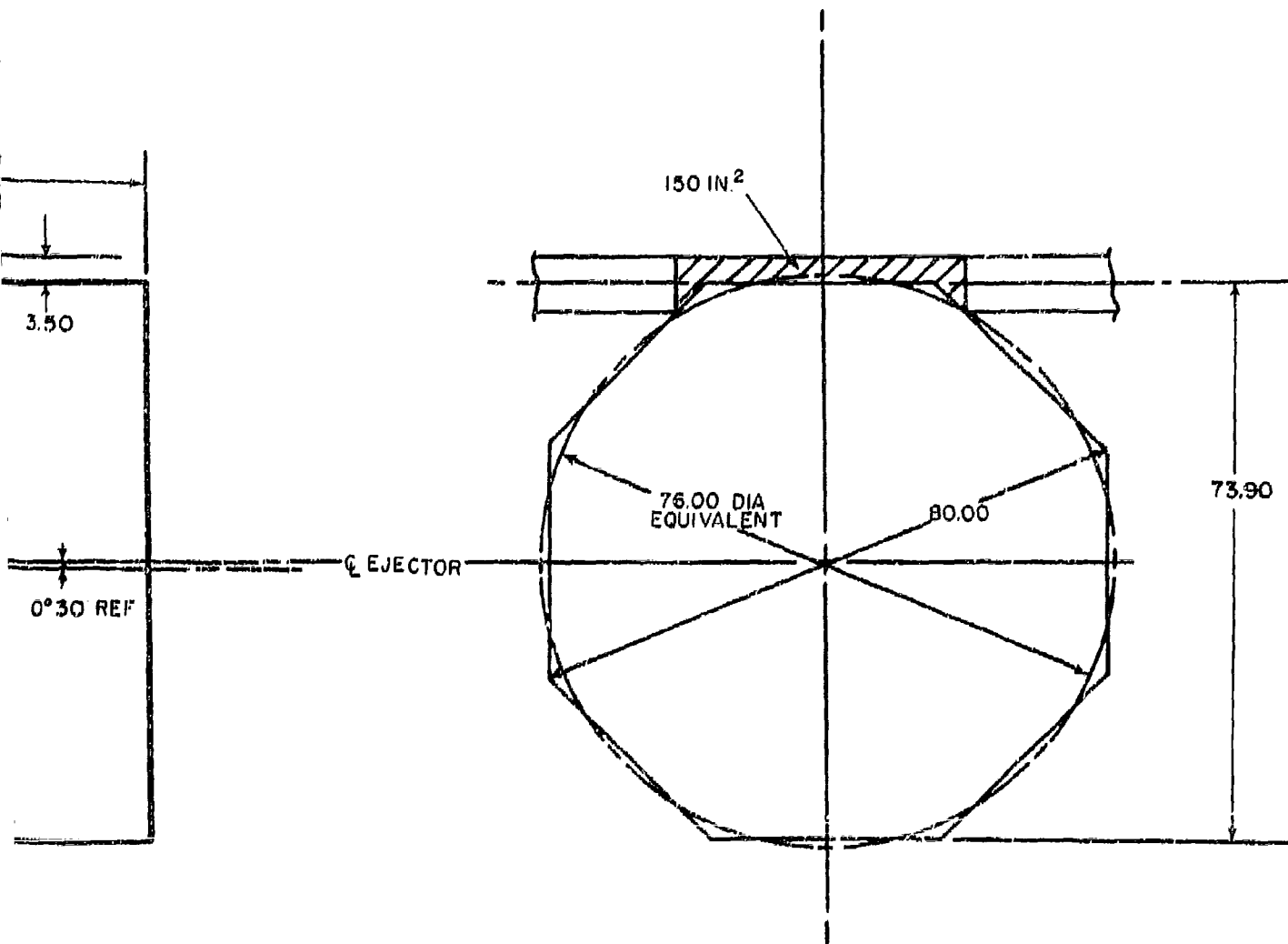
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ARRANGEMENT MOVING REAR ENG. MT AFT  
40.00 & INSERTING ENG. 4.00 INTO WING  
EJECTOR CANTED AT NOZZLE PLANE  
ENG. MOUNT PLANES ARE PARALLEL  
BASE DRAG AREA = 150 in<sup>2</sup>

STF219B 600 LBS./SEC. TURBOFAN

Figure 1-7

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Technical drawing of a wing planform. The drawing shows a rectangular wing with a wavy leading edge on the left. A vertical line on the right is labeled "FRONT END". A horizontal line is labeled "INLET". A vertical dimension of 29.60 is indicated. A horizontal dimension of 3.10 is indicated. A vertical dimension of 5" is indicated. A line is labeled "WING PIVOT PLANE".

3.10 -

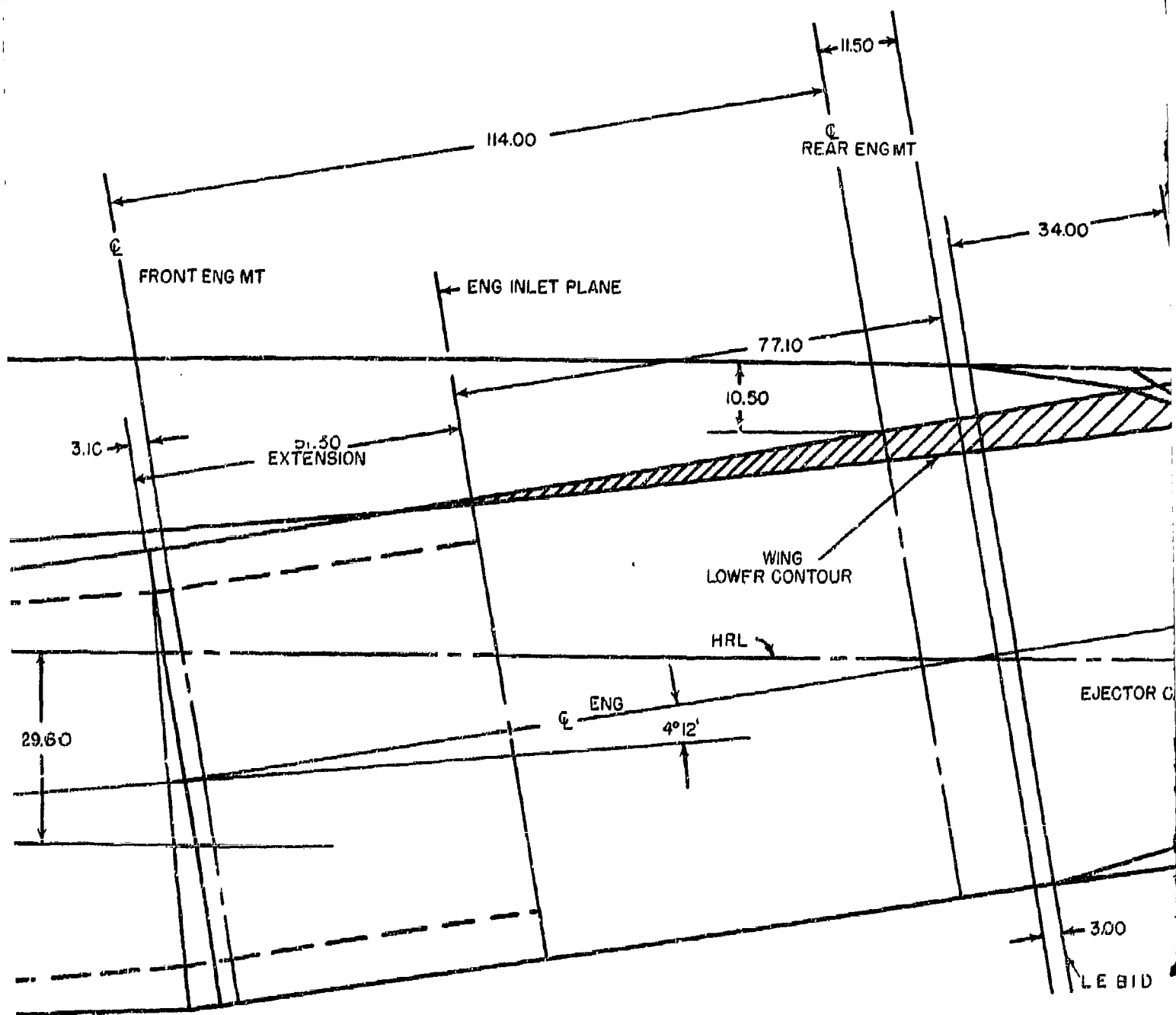
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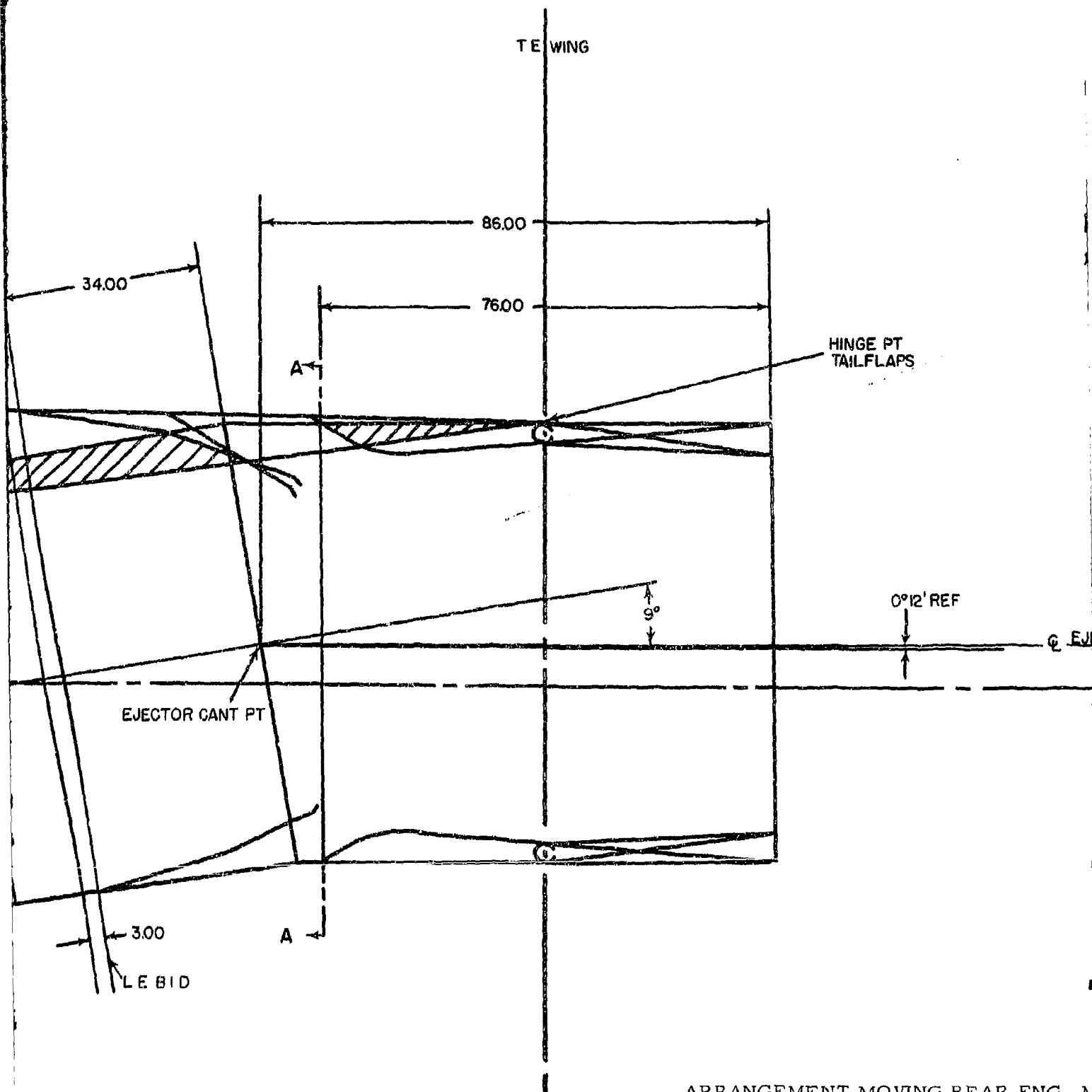
INLET

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WING PIVOT  
PLANE

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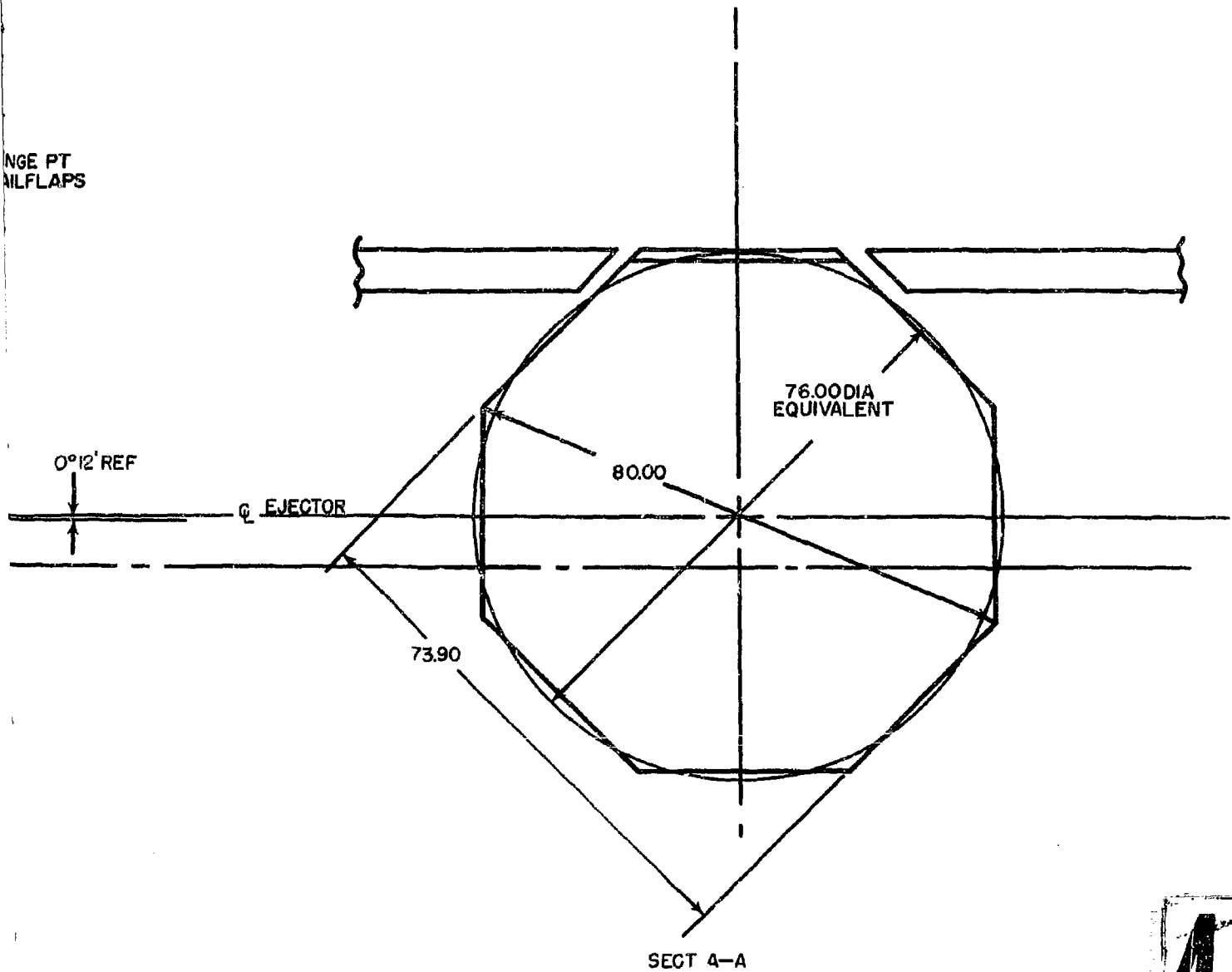




ARRANGEMENT MOVING REAR ENG. M  
40.00 AFT & SETTING EJECTOR TAIL F  
AT T. E. OF WING  
EJECTOR CANTED AT NOZZLE PLANE

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MOVING REAR ENG. MT.  
EJECTOR TAIL FLAPS  
NG  
ED AT NOZZLE PLANE

STF219B 600 LBS./SEC. TURBOFAN

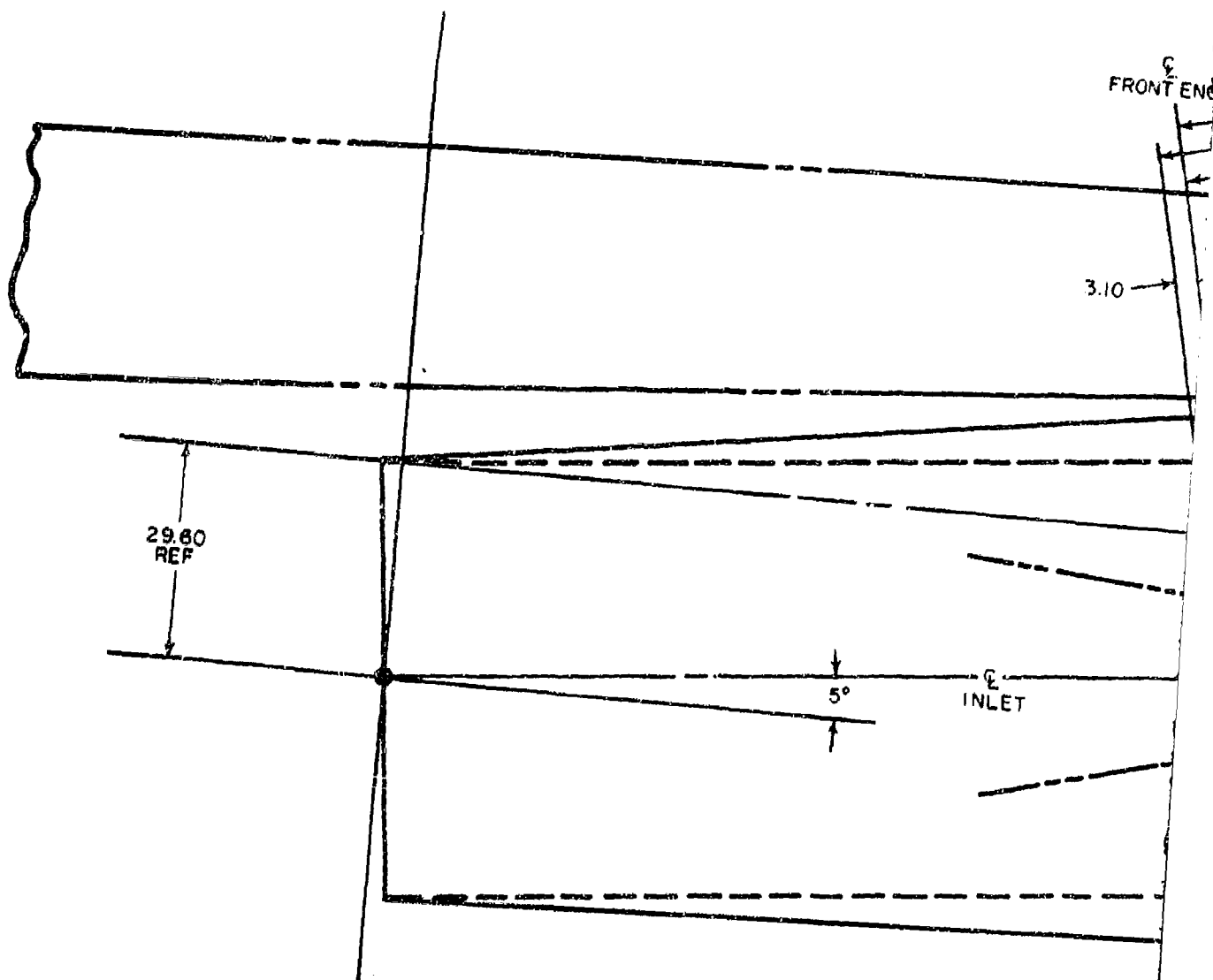
Figure 1-8

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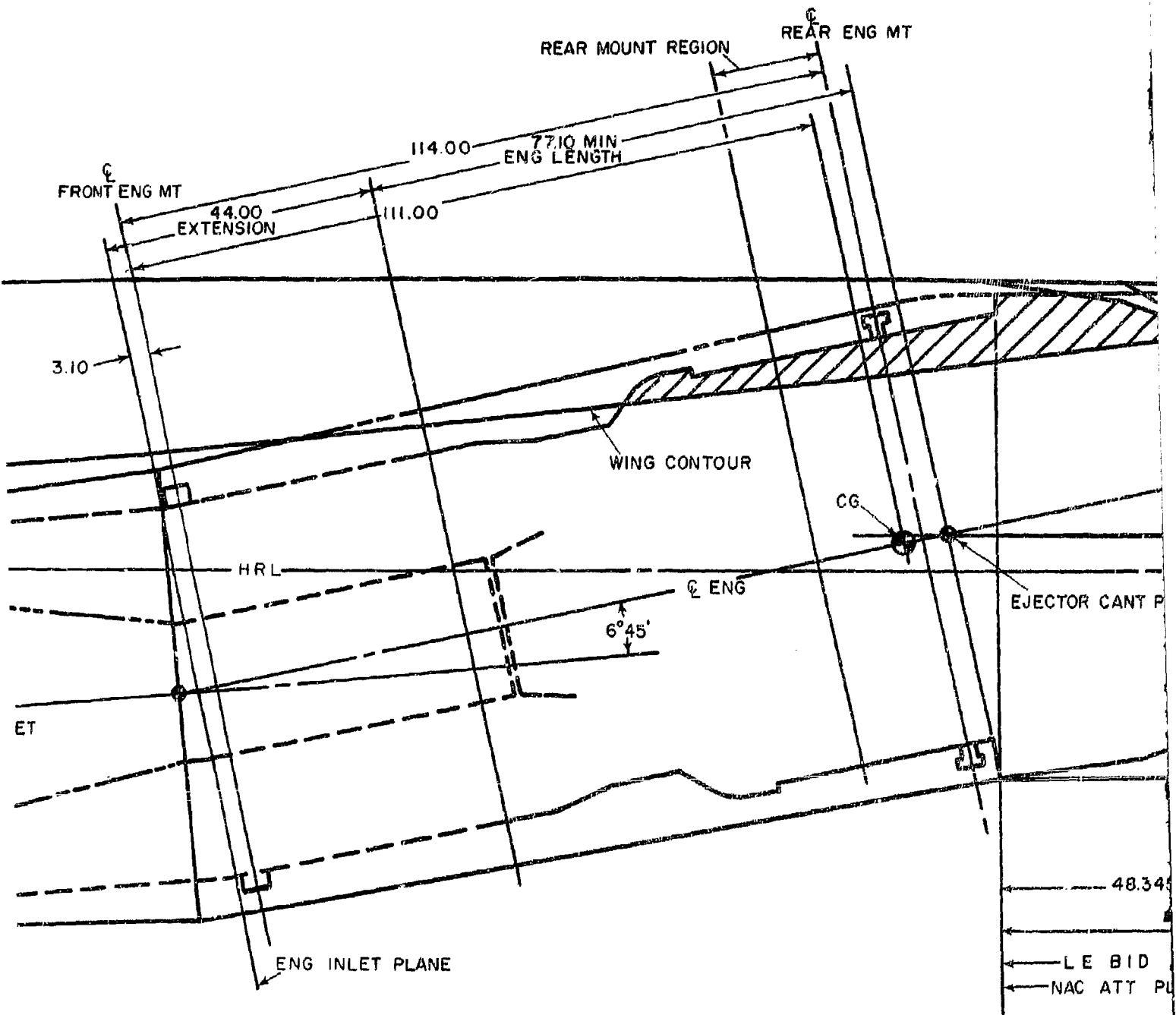
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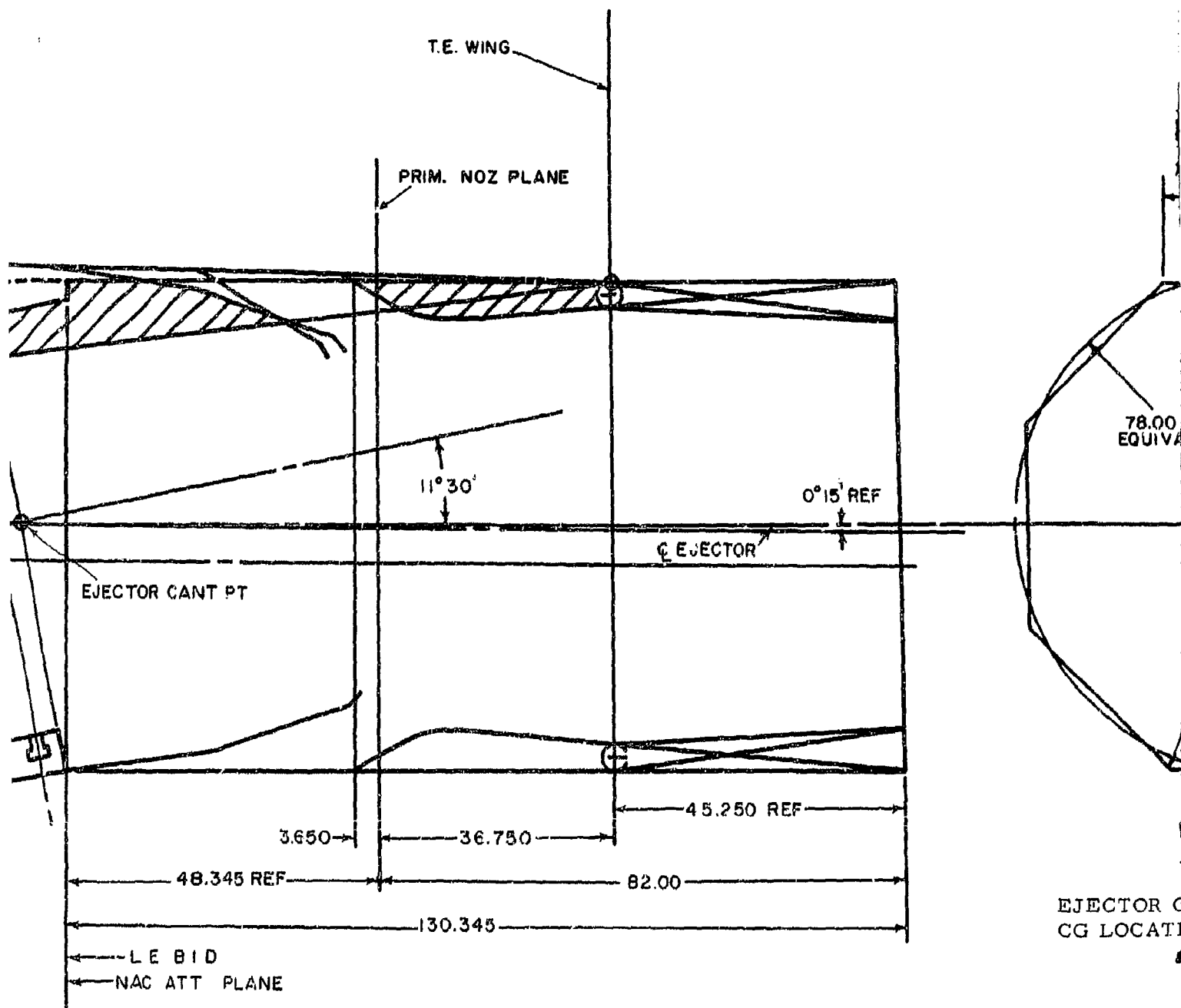


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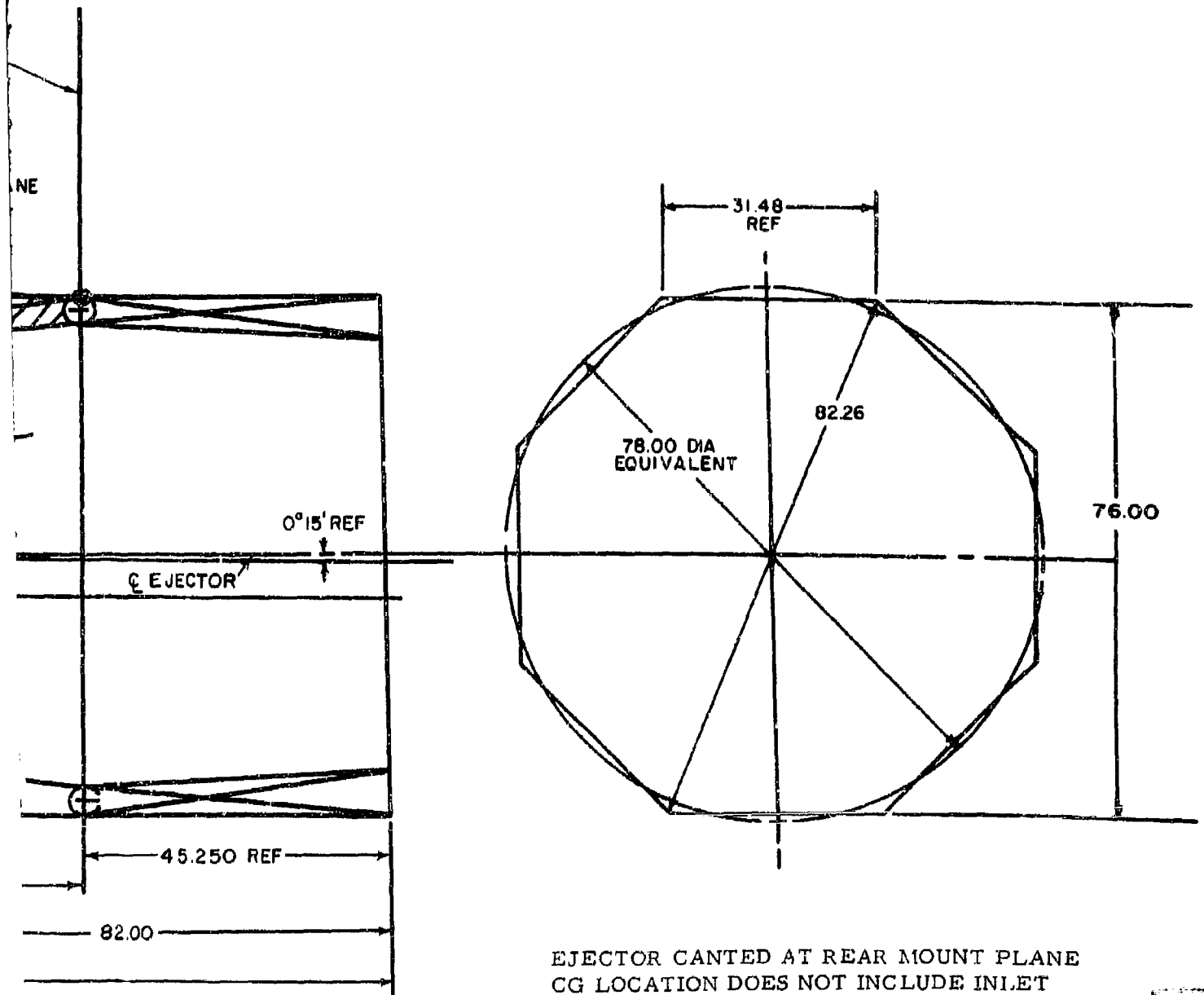
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PWA-2600



STF219B 600 LBS./SEC. TURBOFAN

Figure 1-9

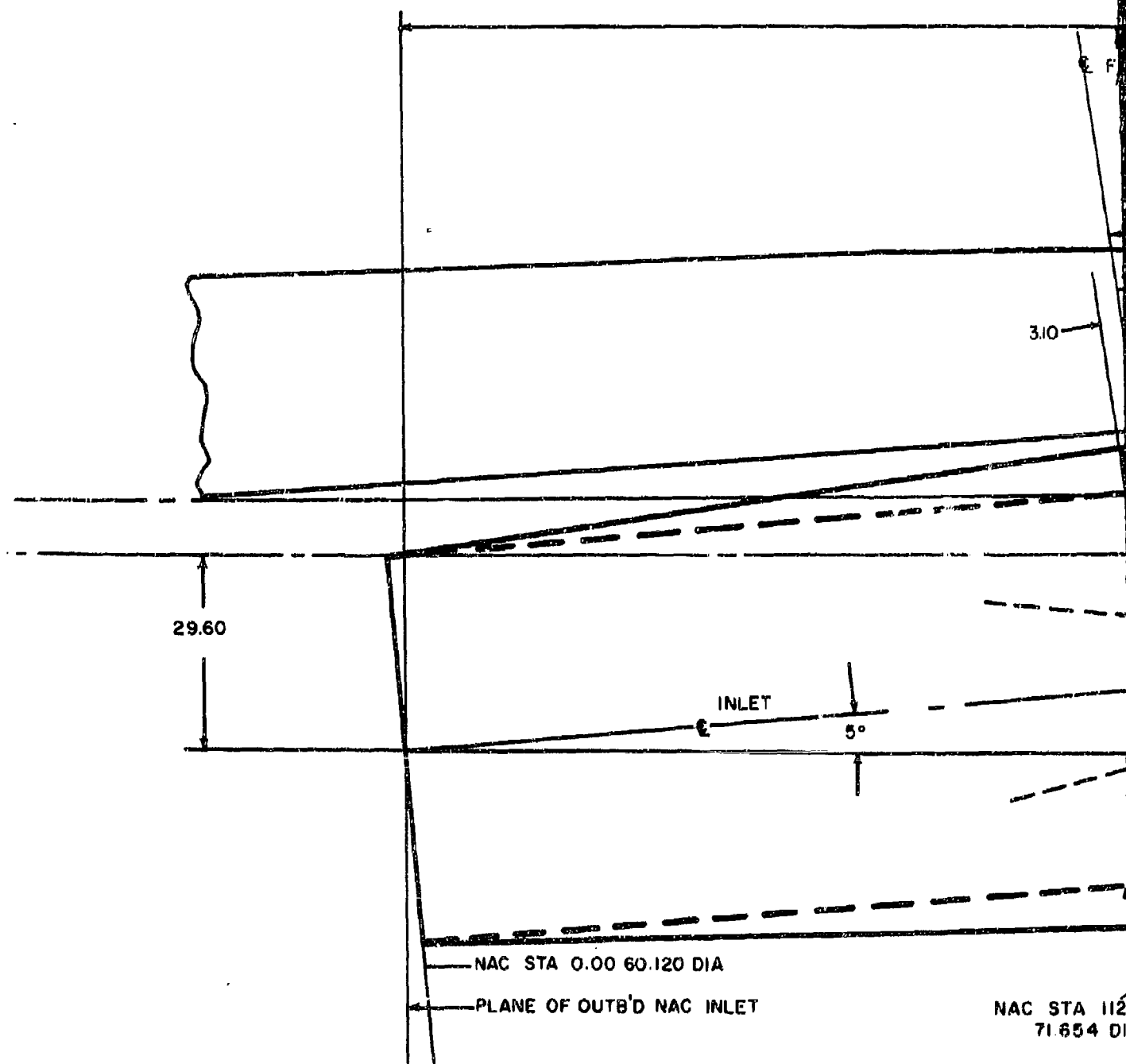
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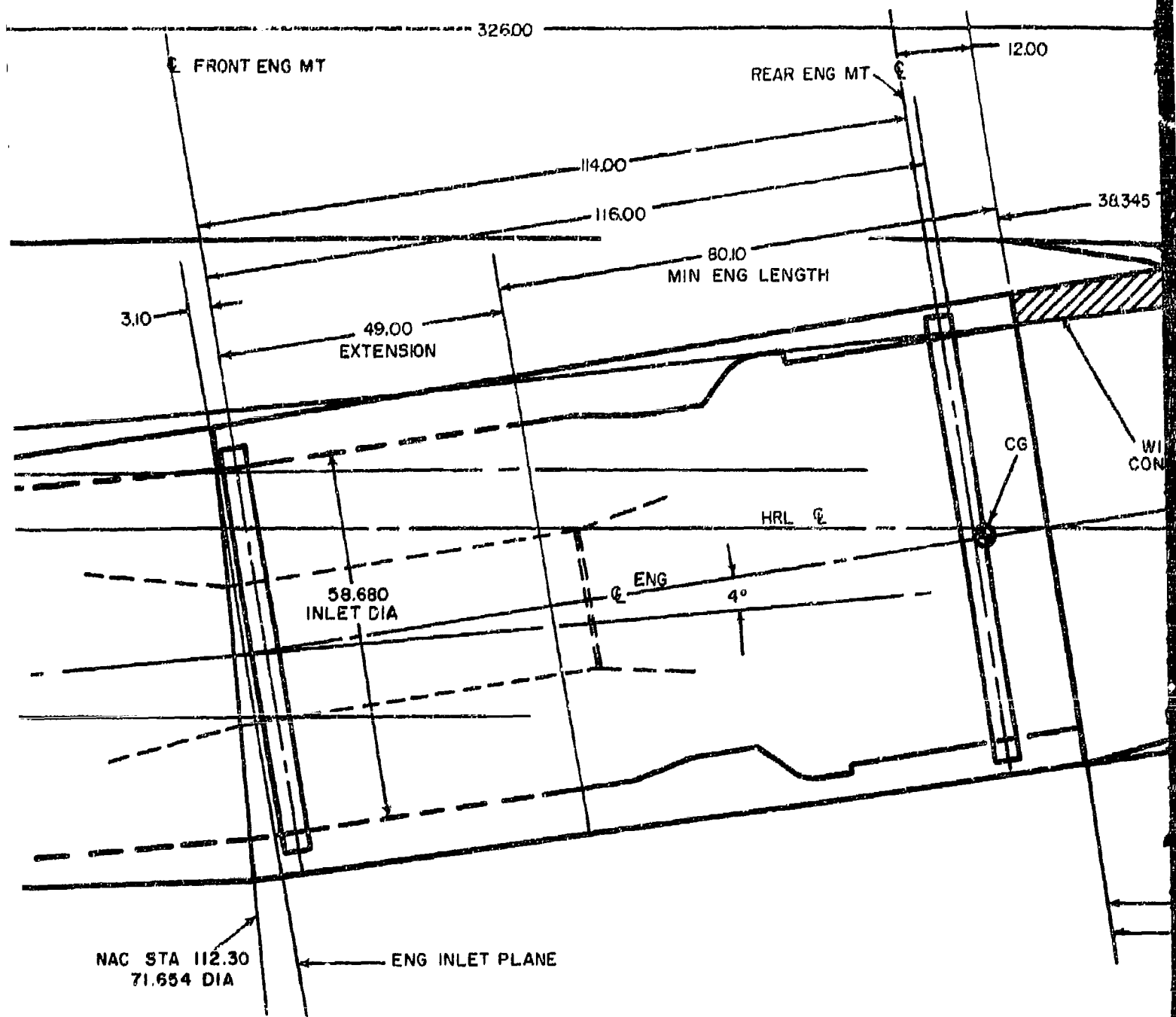
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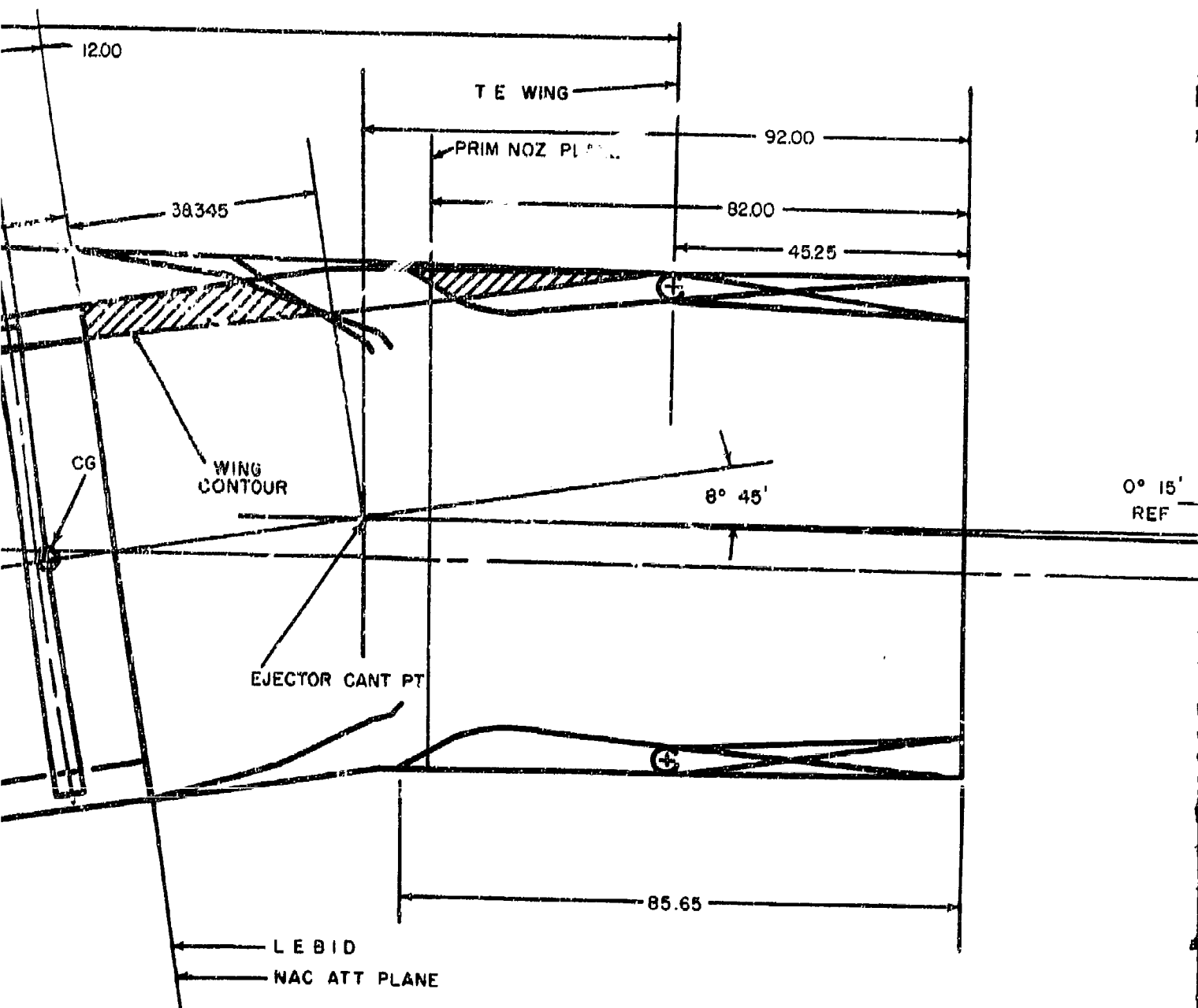
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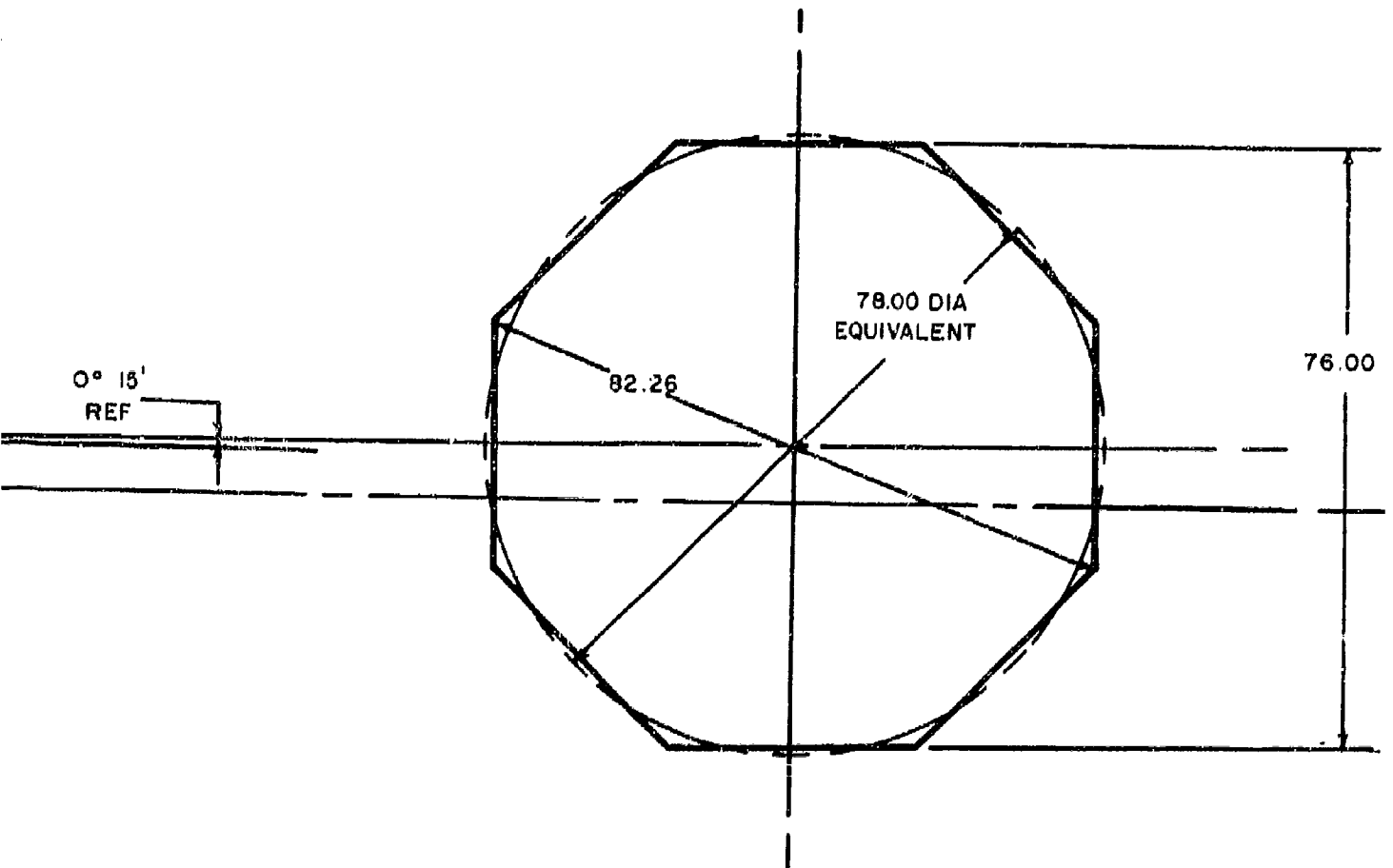






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EJECTOR CANTED 10.00 FWD. OF PRIMARY  
NOZZLE PLANE  
CG LOCATION DOES NOT INCLUDE INLET

STF219B 600 LBS./SEC. TURBOFAN

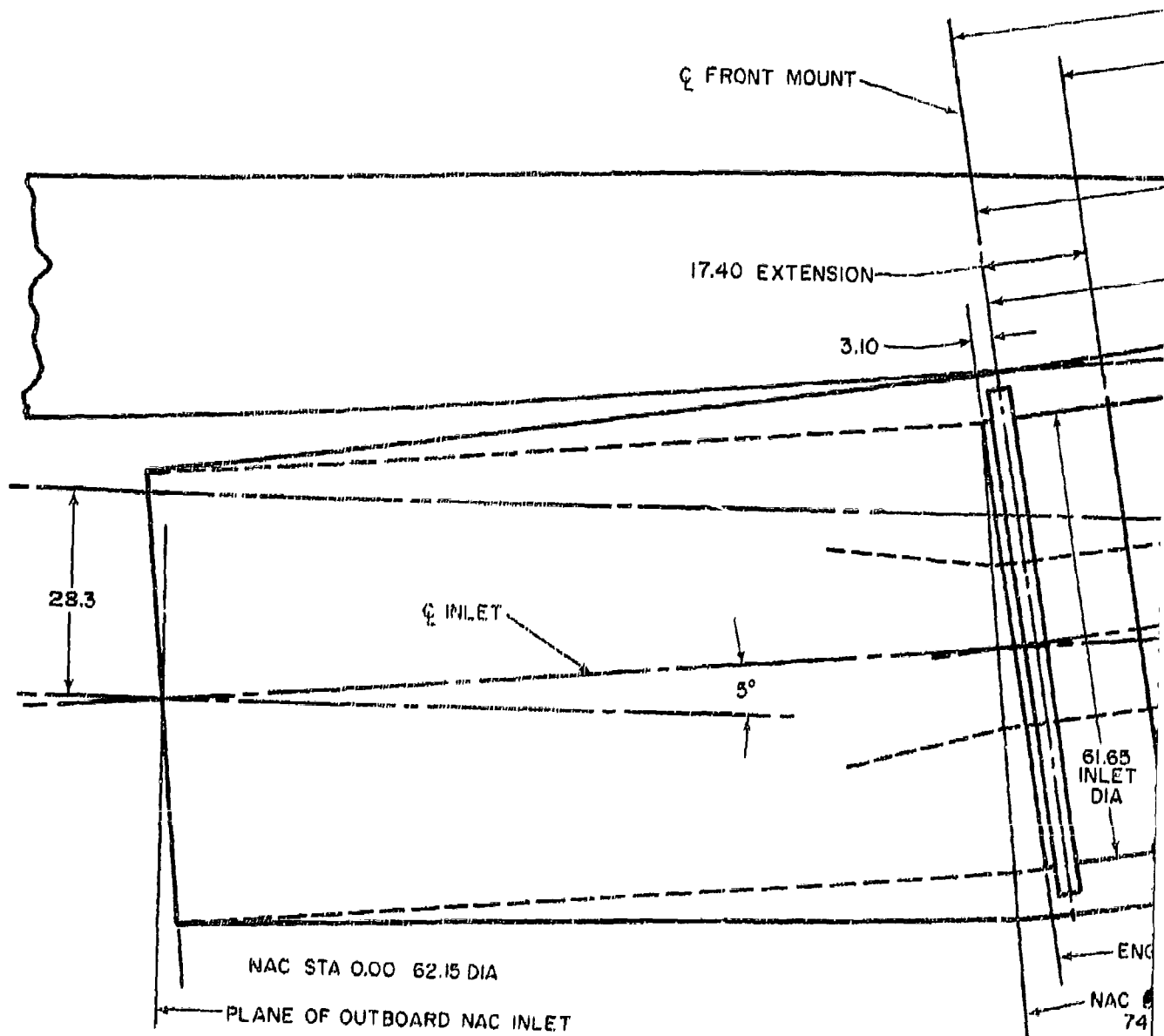
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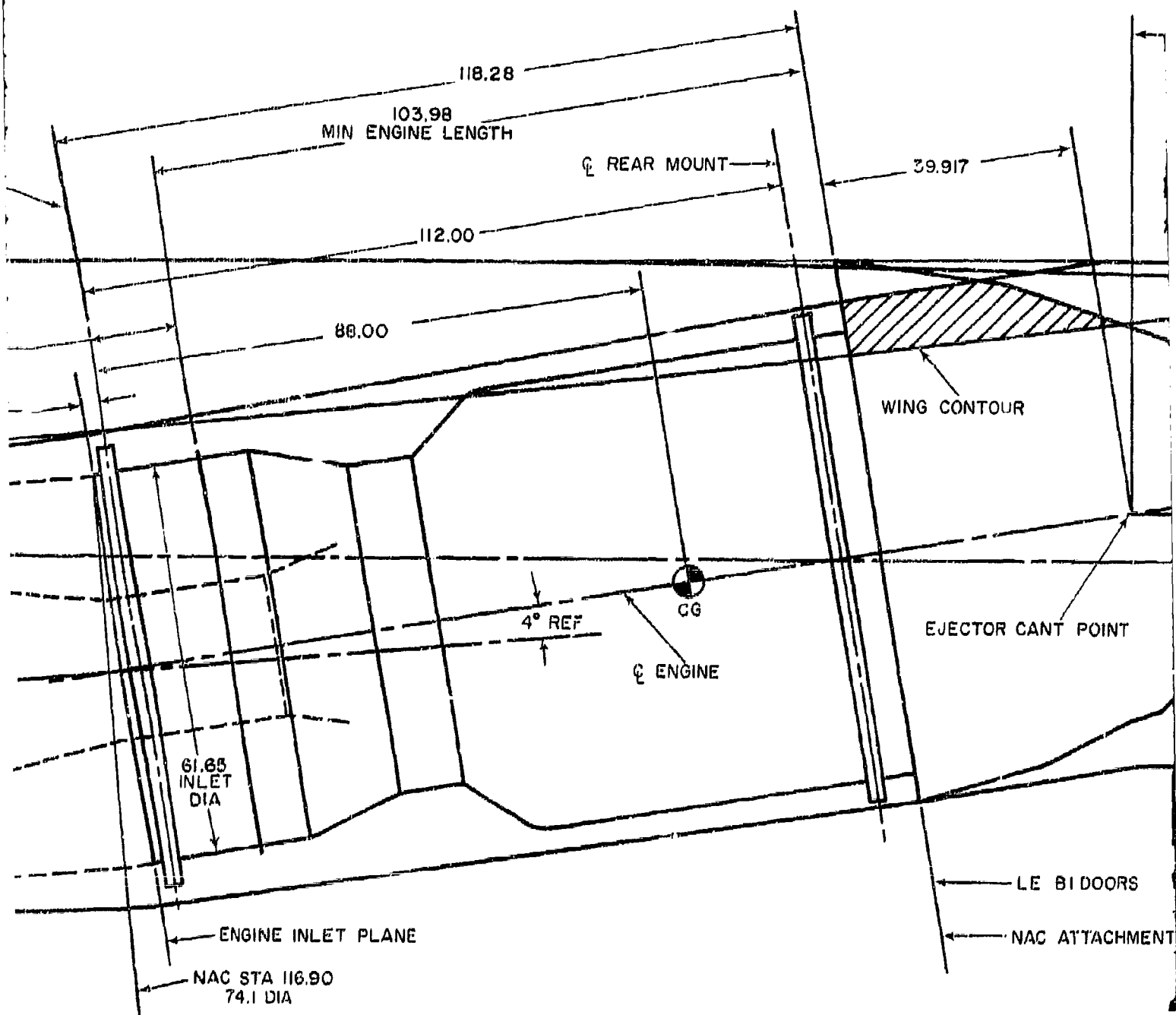
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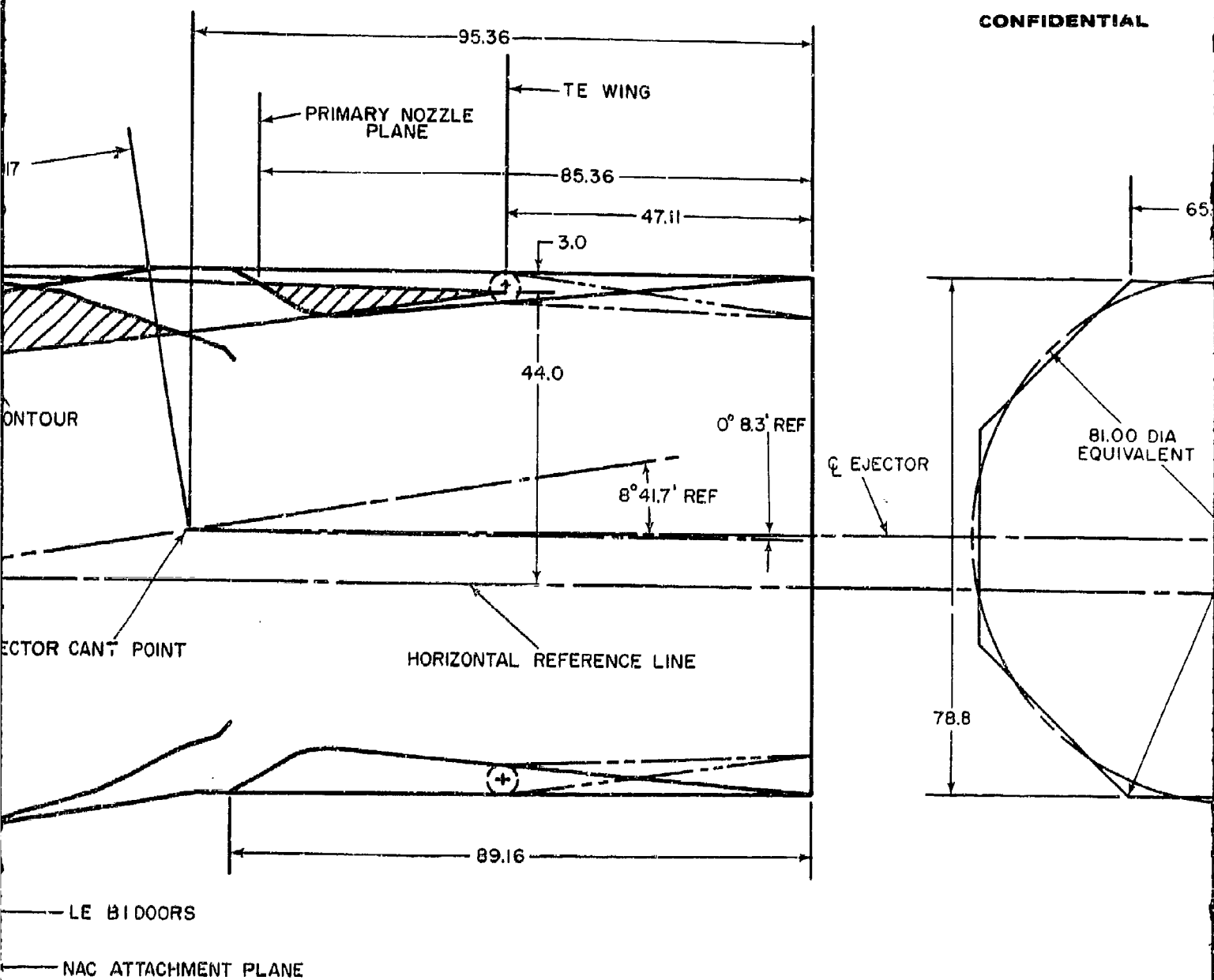
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CG LOCATION DOES NOT INCLUDE INLET

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CG LOCATION

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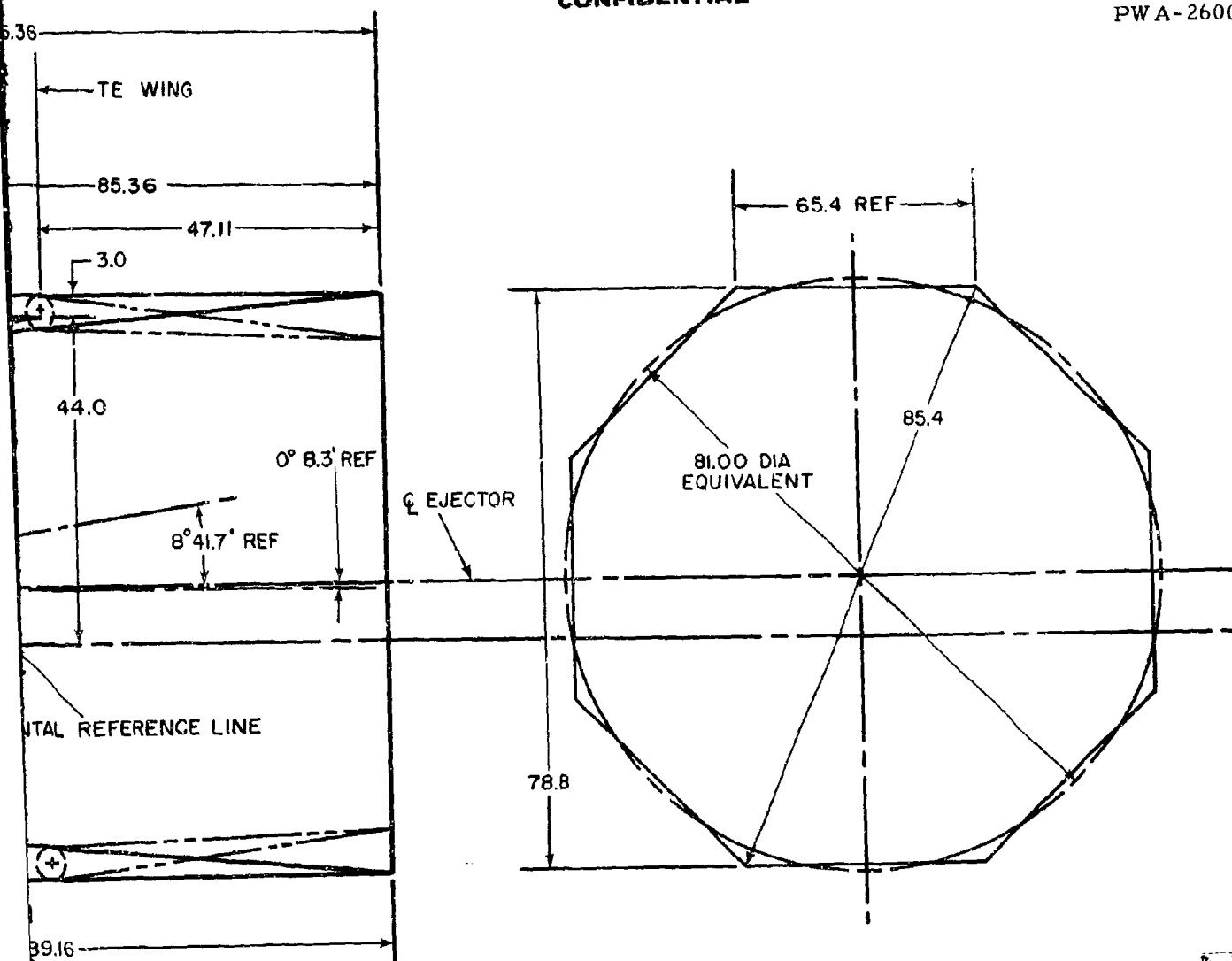
Figure 1-11

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CG LOCATION DOES NOT INCLUDE INLET

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NOZZLE PLANE  
CG LOCATION DOES NOT INCLUDE INLET

STF219 650 LBS./SEC. TURBOFAN

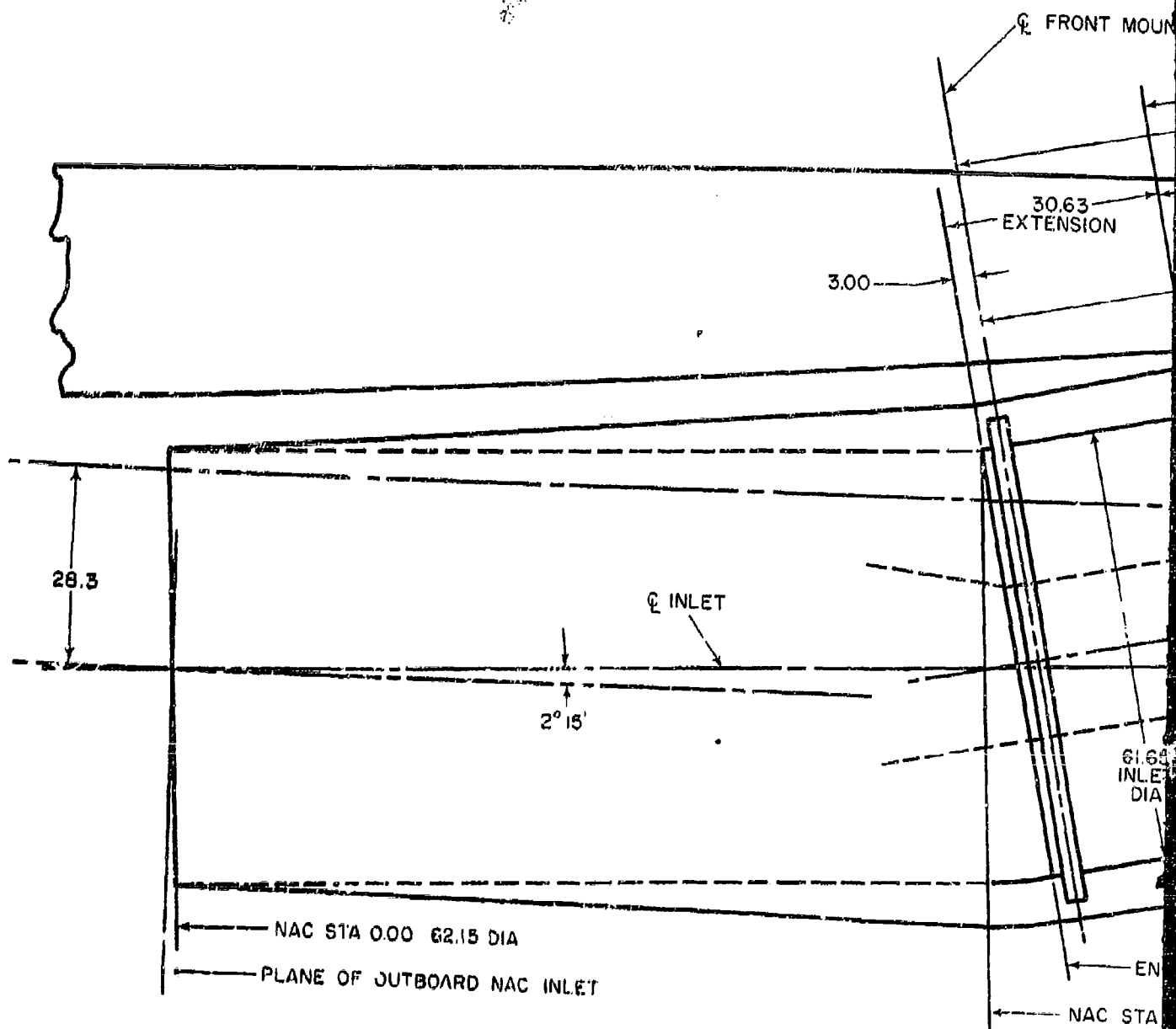
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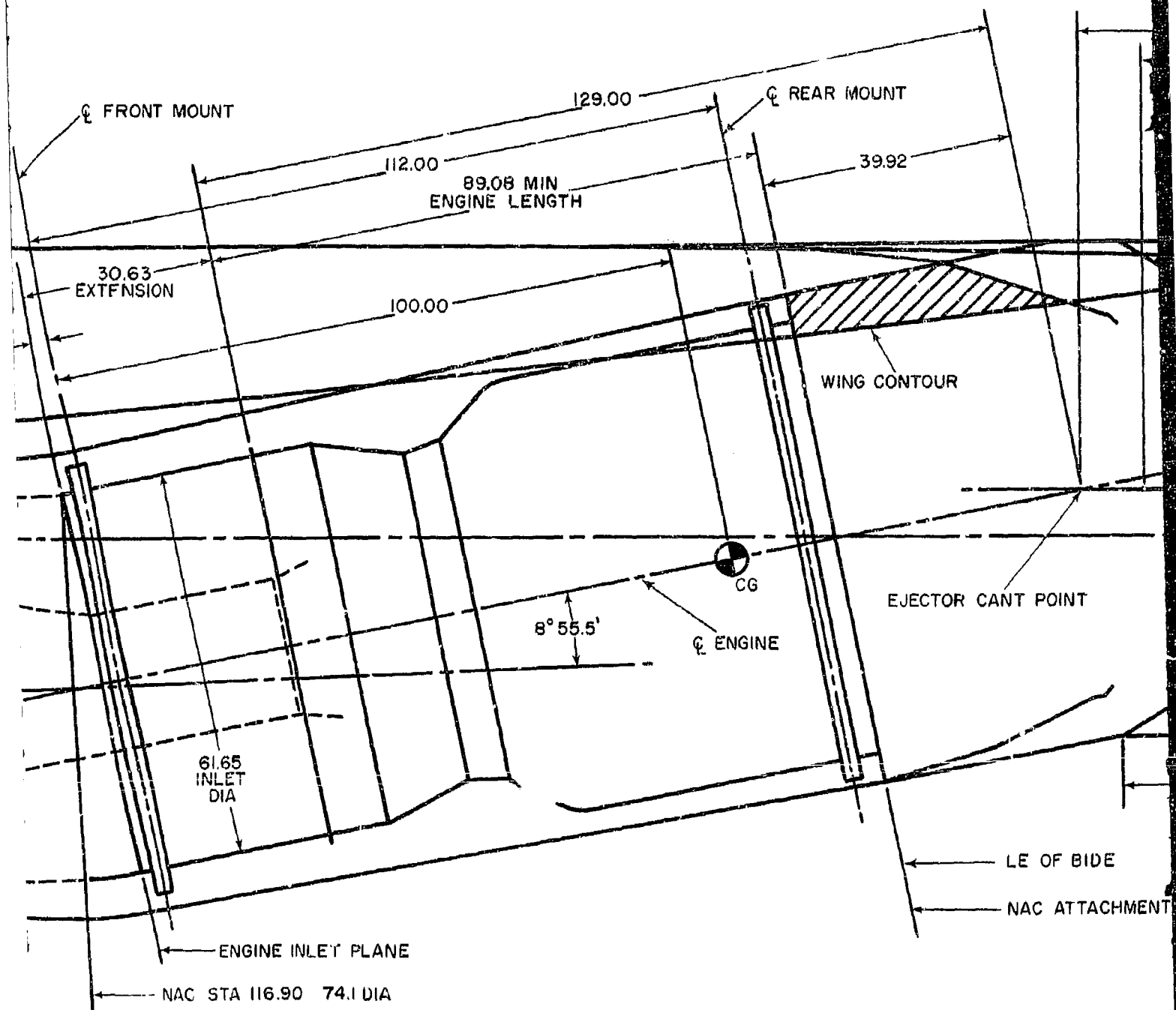
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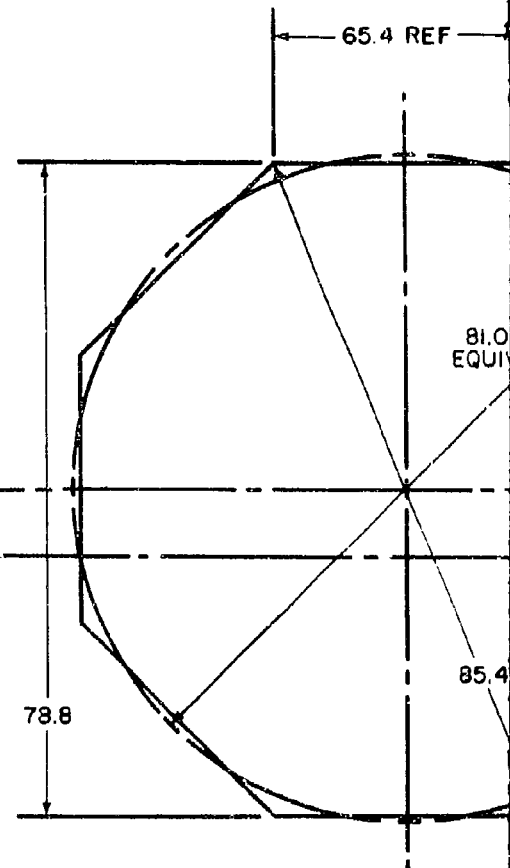
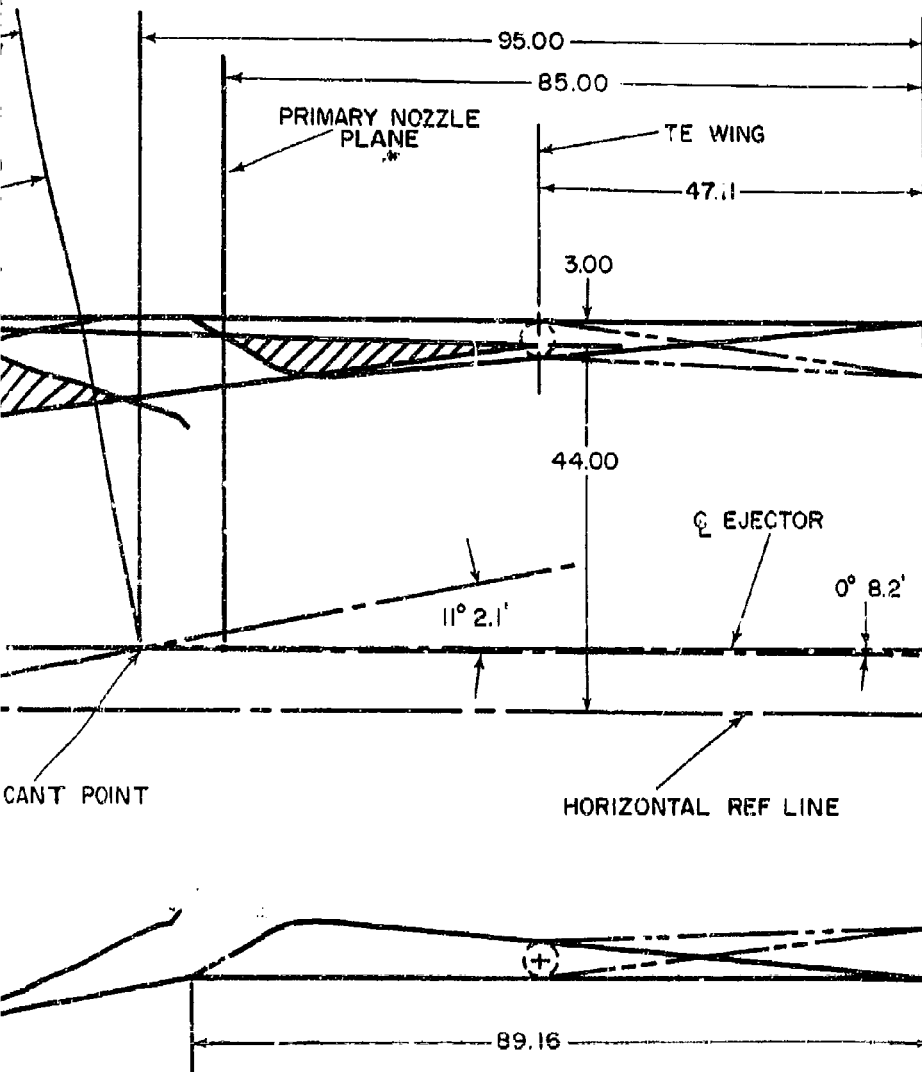
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EJECTOR CANTED 10.00 FWD.  
NOZZLE PLANE  
CG LOCATION DOES NOT INCLUDE INLET

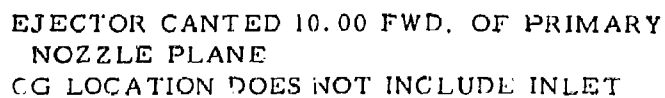
CG LOCATION DOES NOT INCLUDE INLET

STF219 650 LBS./SEC. TURBOFAN

Figure 1-12

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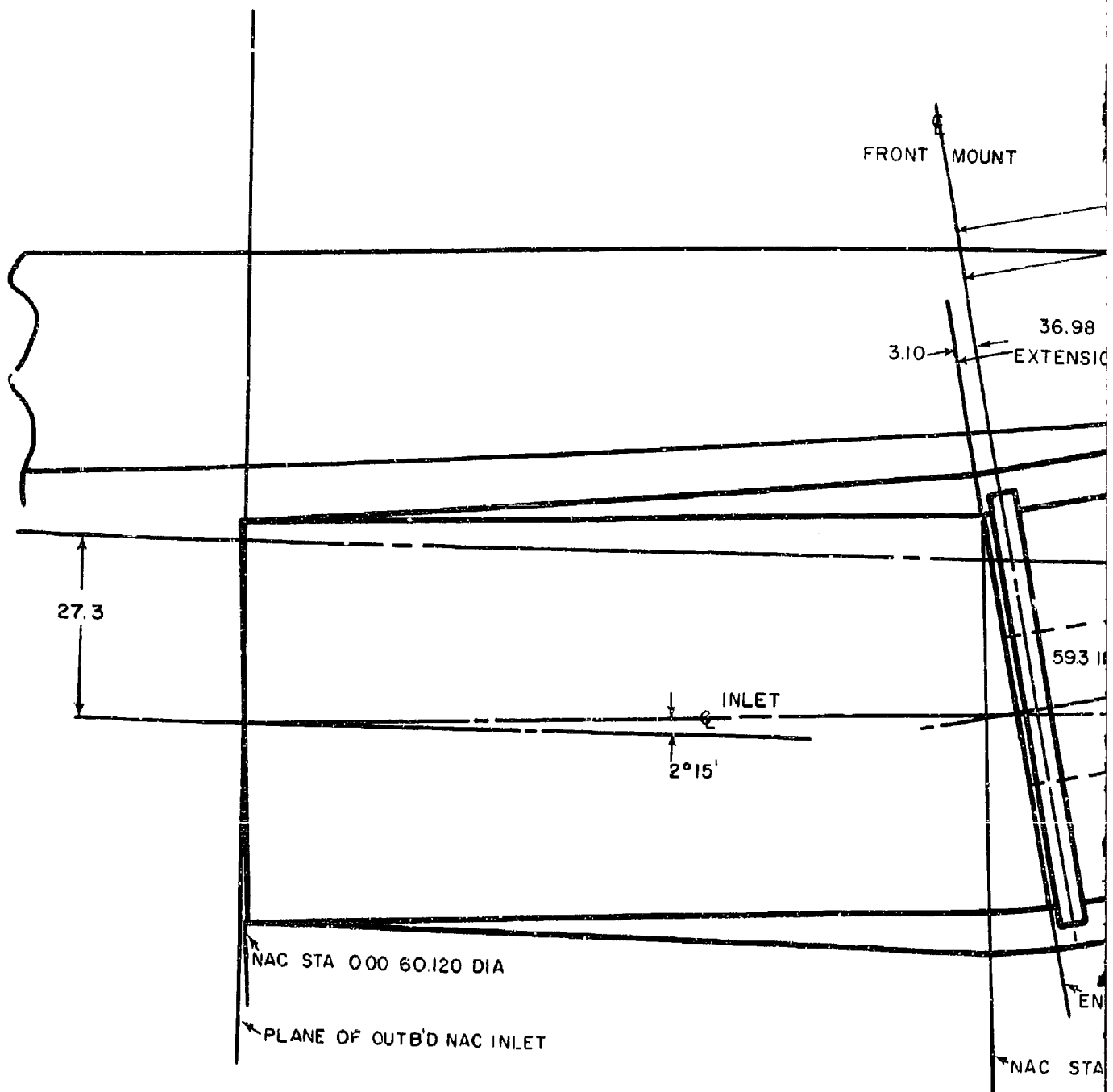


STF219 650 LBS. /SEC. TURBOFAN

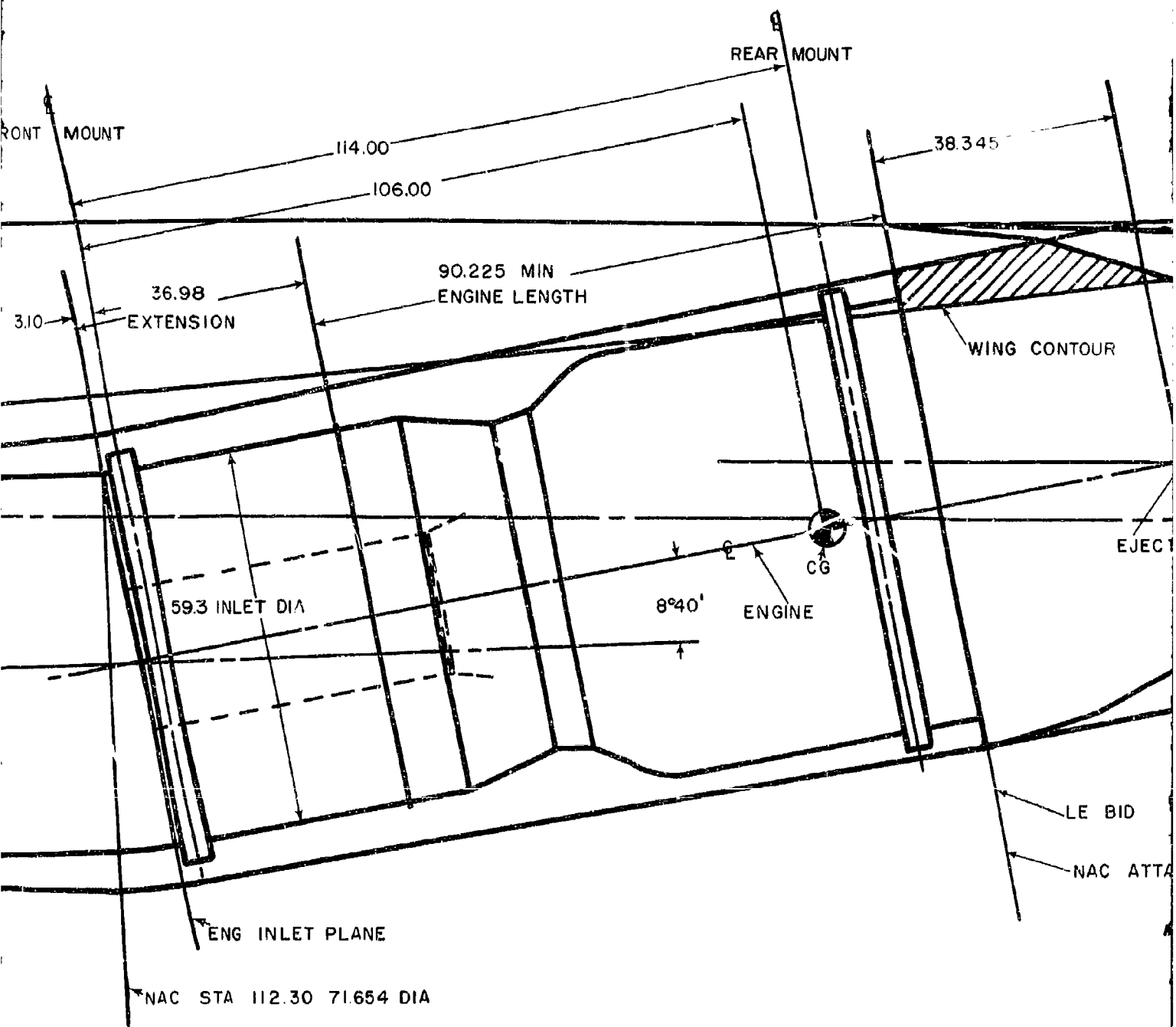
Figure 1-12

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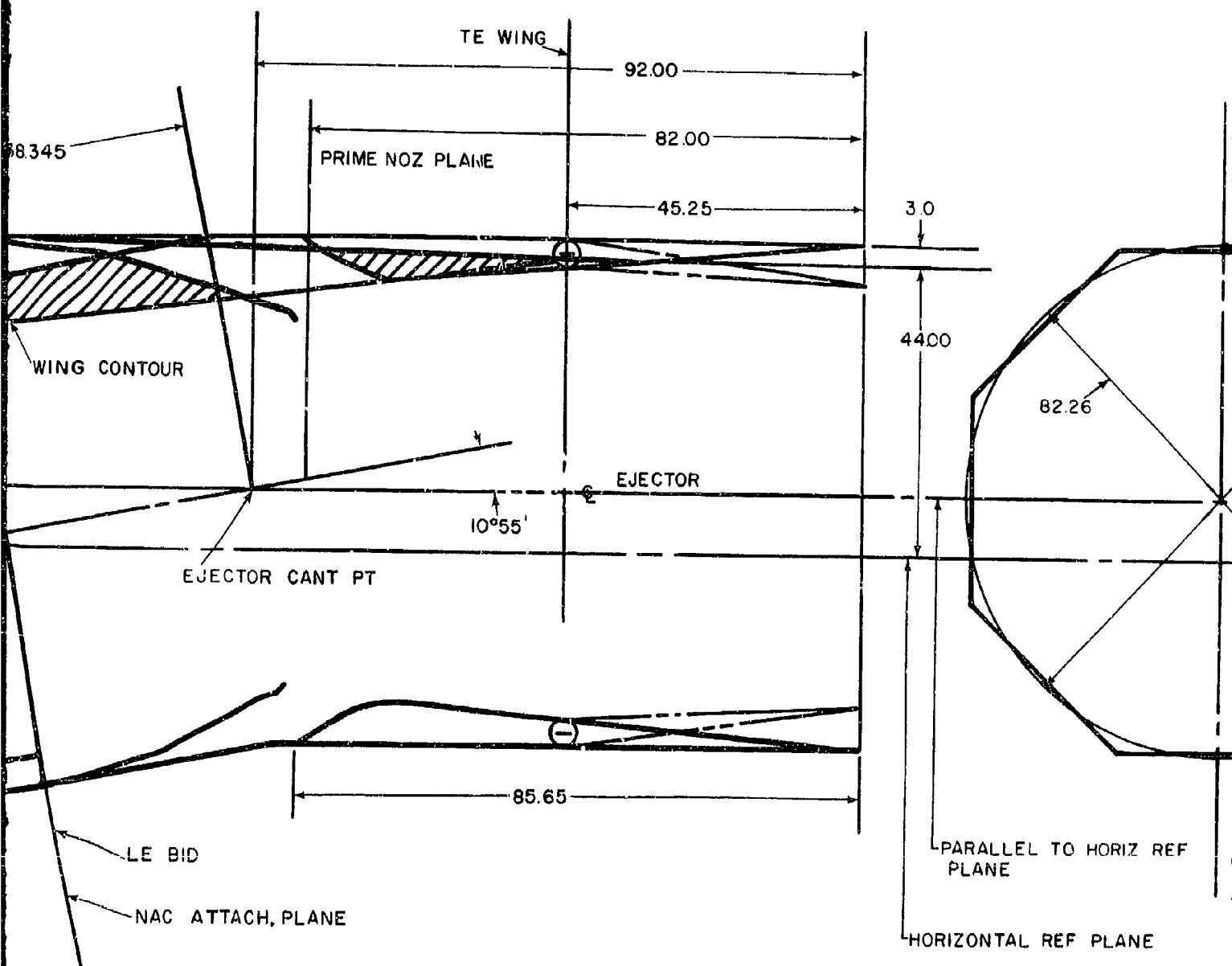
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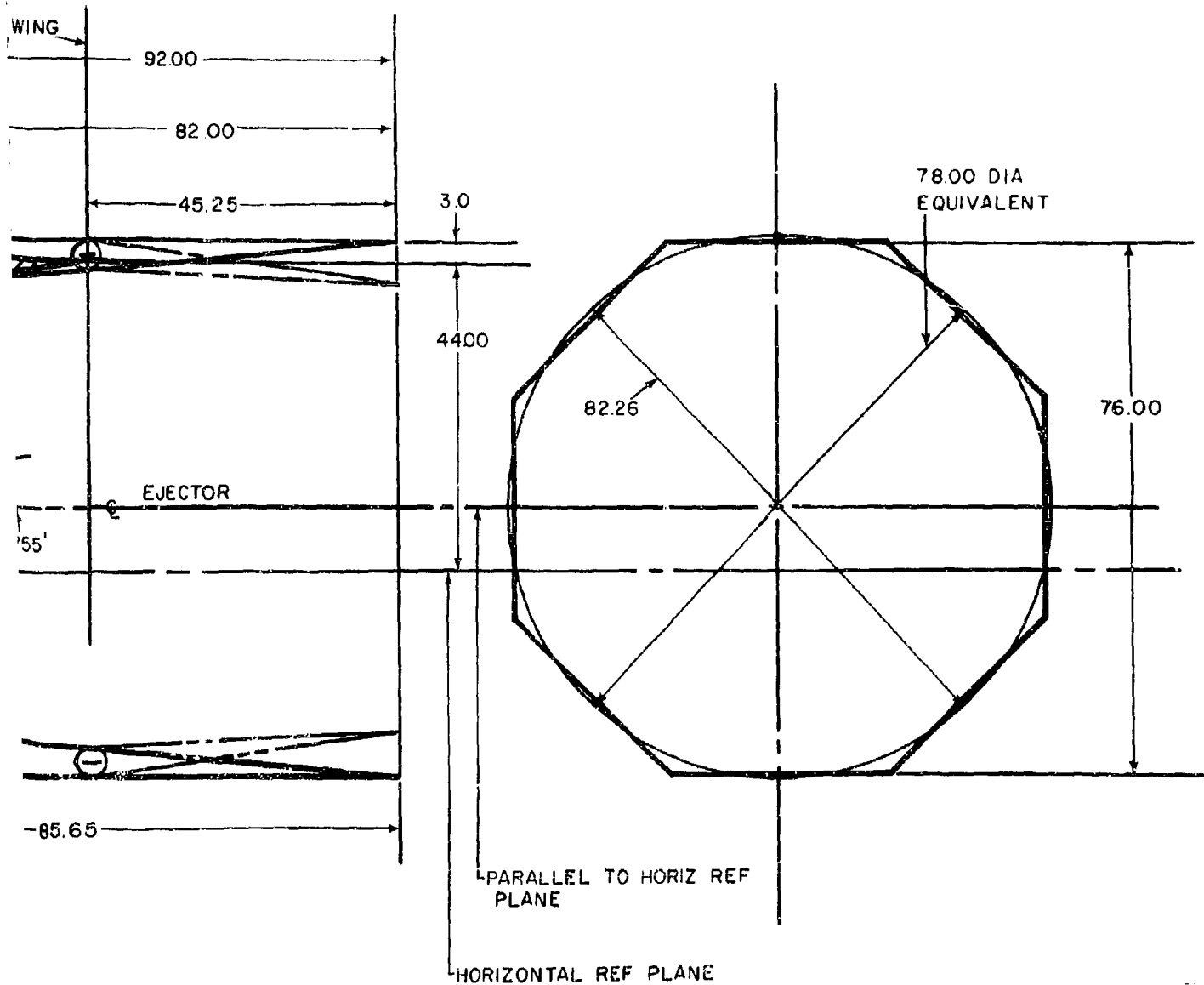
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Figure 1-13

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Figure 1-13

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REPLACEMENT AT 3 YEAR INTERVALS  
REPLACEMENT AFTER 12 YEARS  
FROM 1960-1970

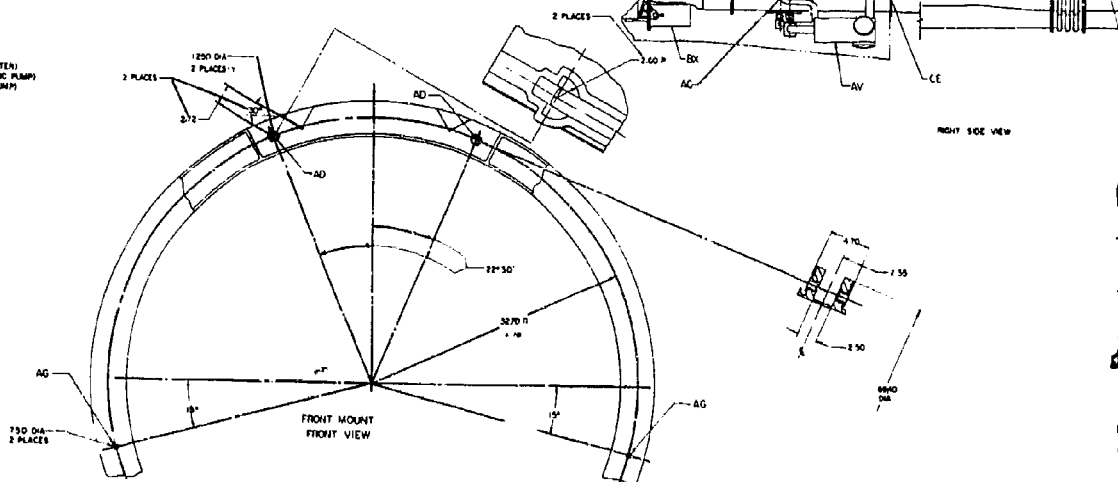
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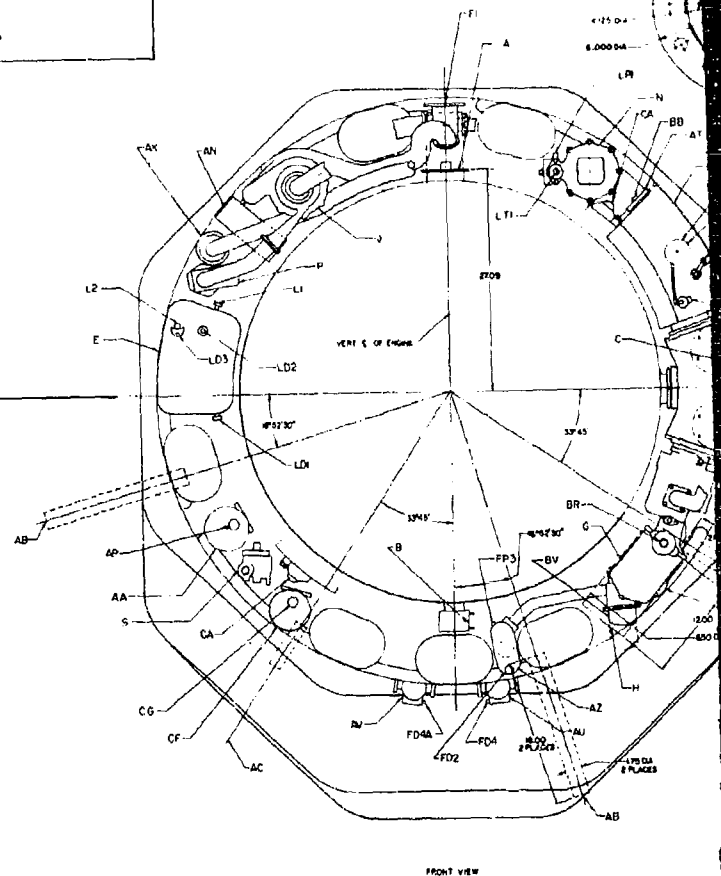
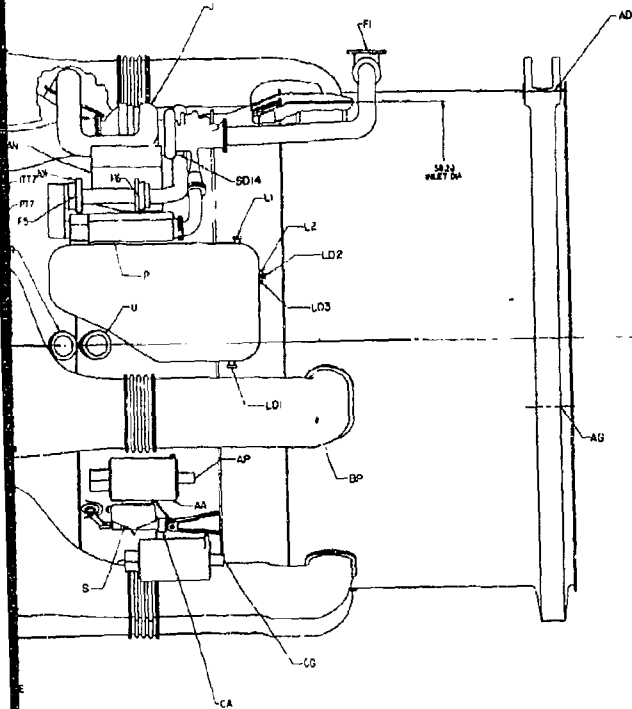
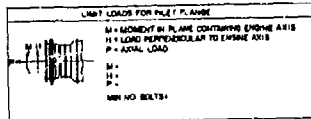
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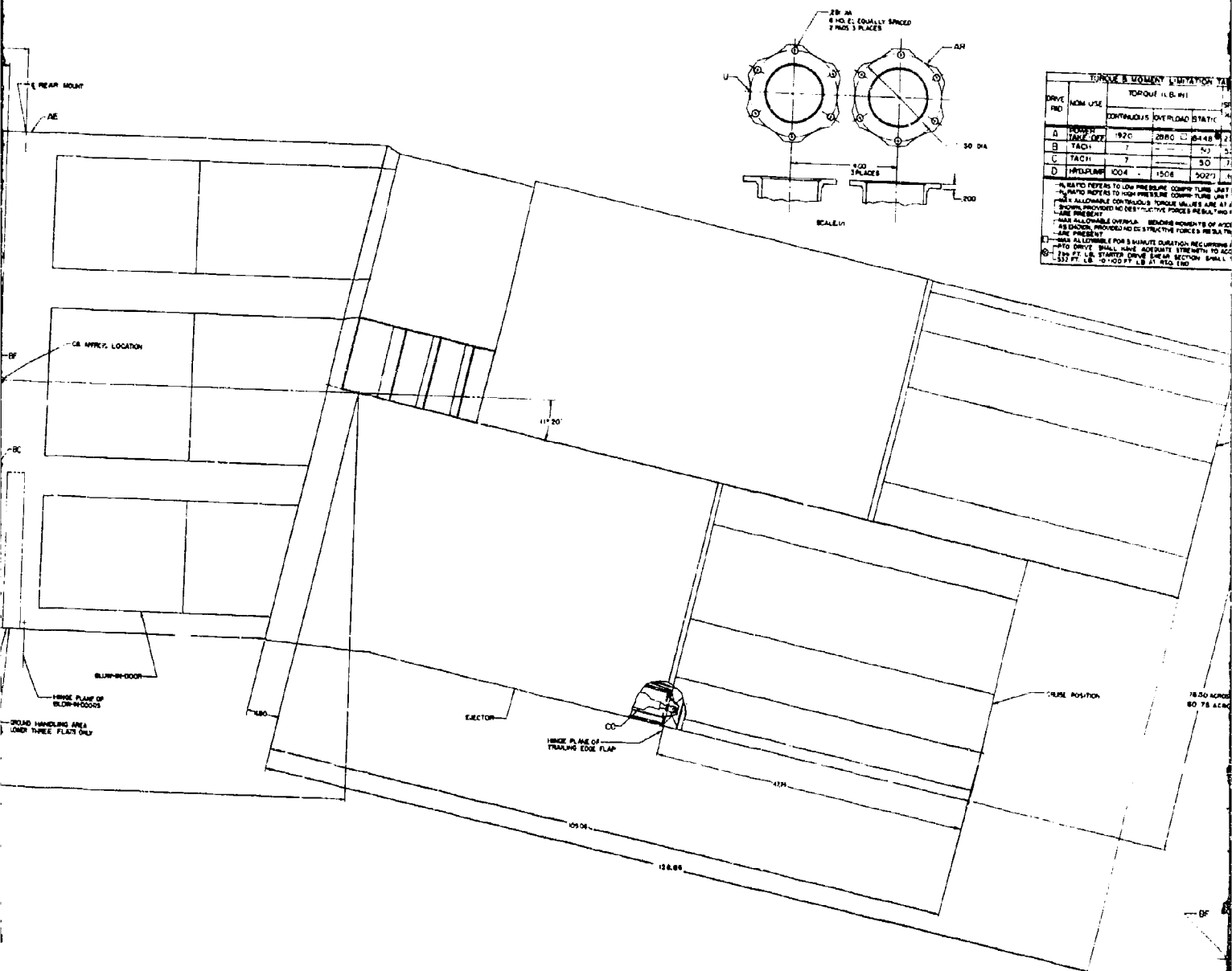
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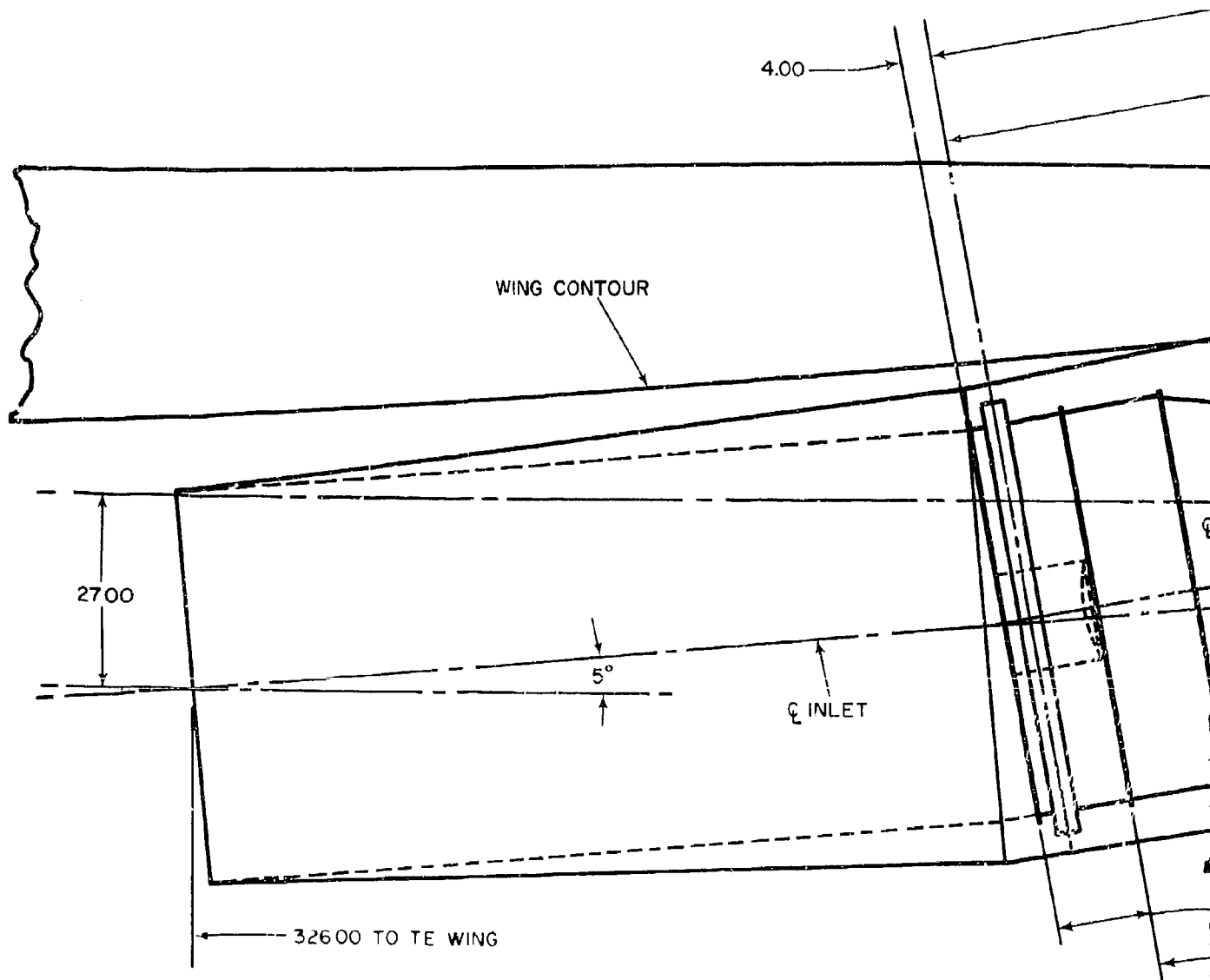


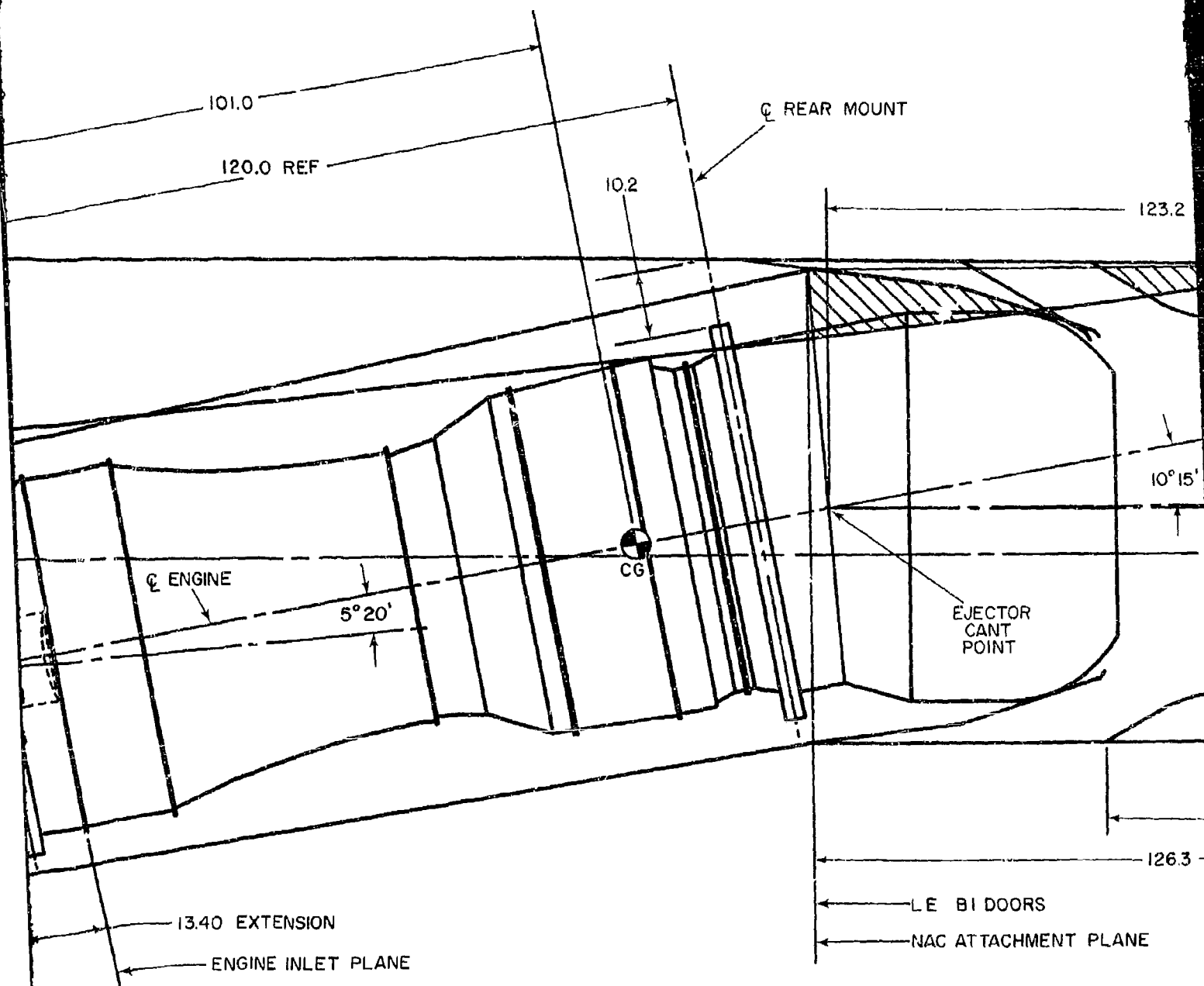
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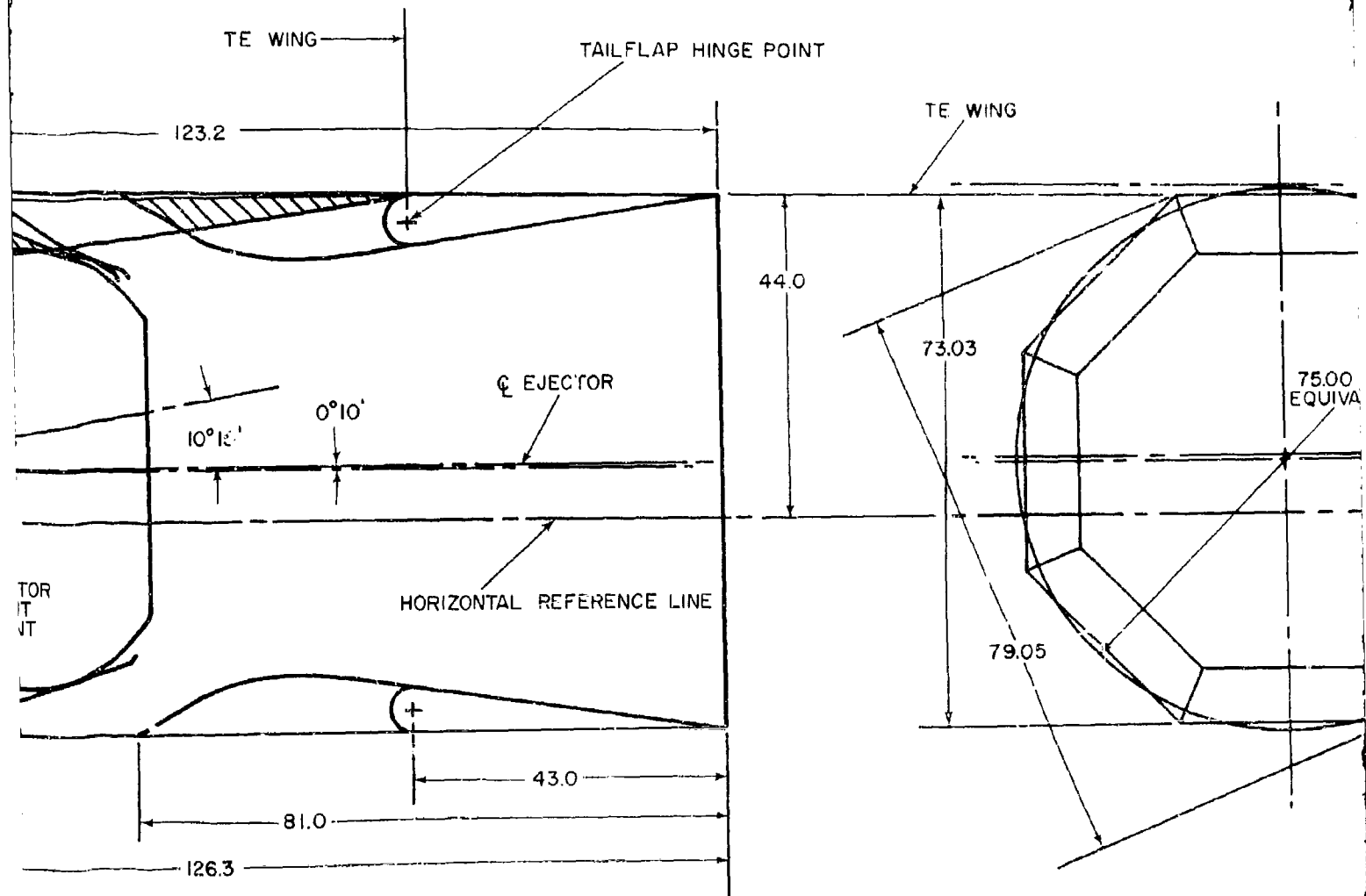






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CG LOCATION DOES NOT INCLUDE THE INLET

F MAX. TURBINE INLET TEMP.  
 TOE CANTED AT REAR MOUNT PLANE  
 LOCATION DOES NOT INCLUDE THE INLET

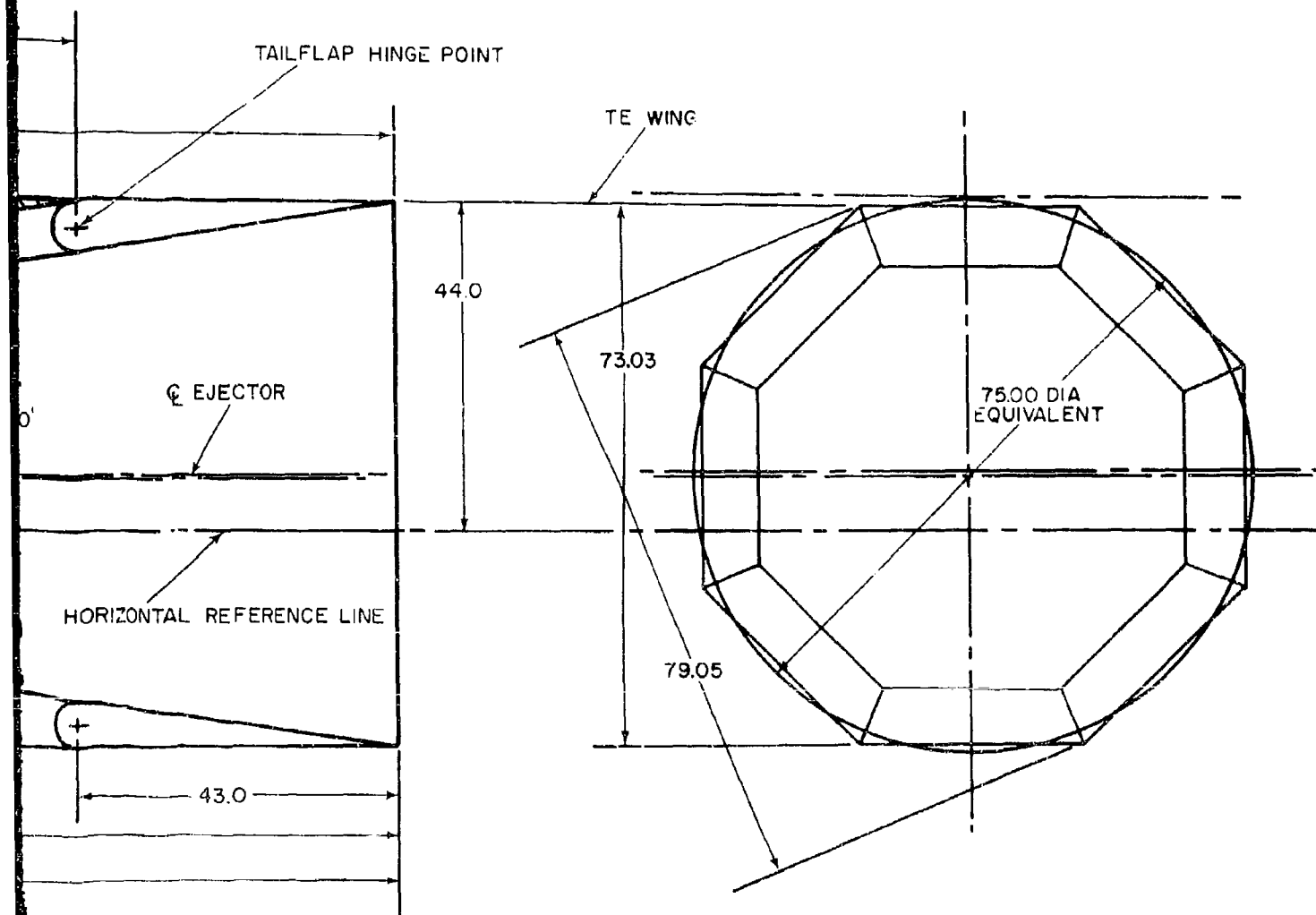
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Figure 1-15

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CG LOCATION DOES NOT INCLUDE THE INLET

TEMP.  
MOUNT PLANE  
SIDE THE INLET

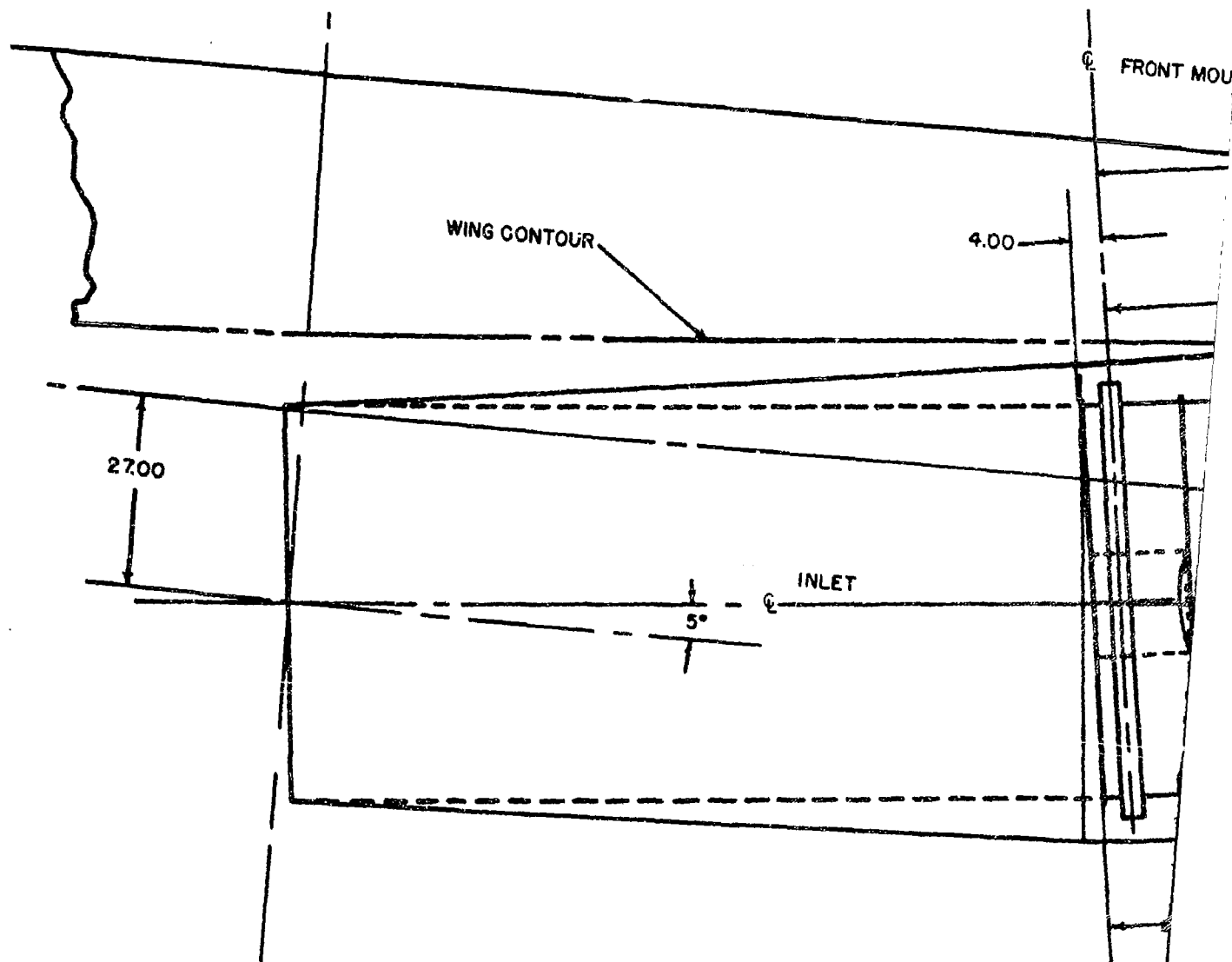
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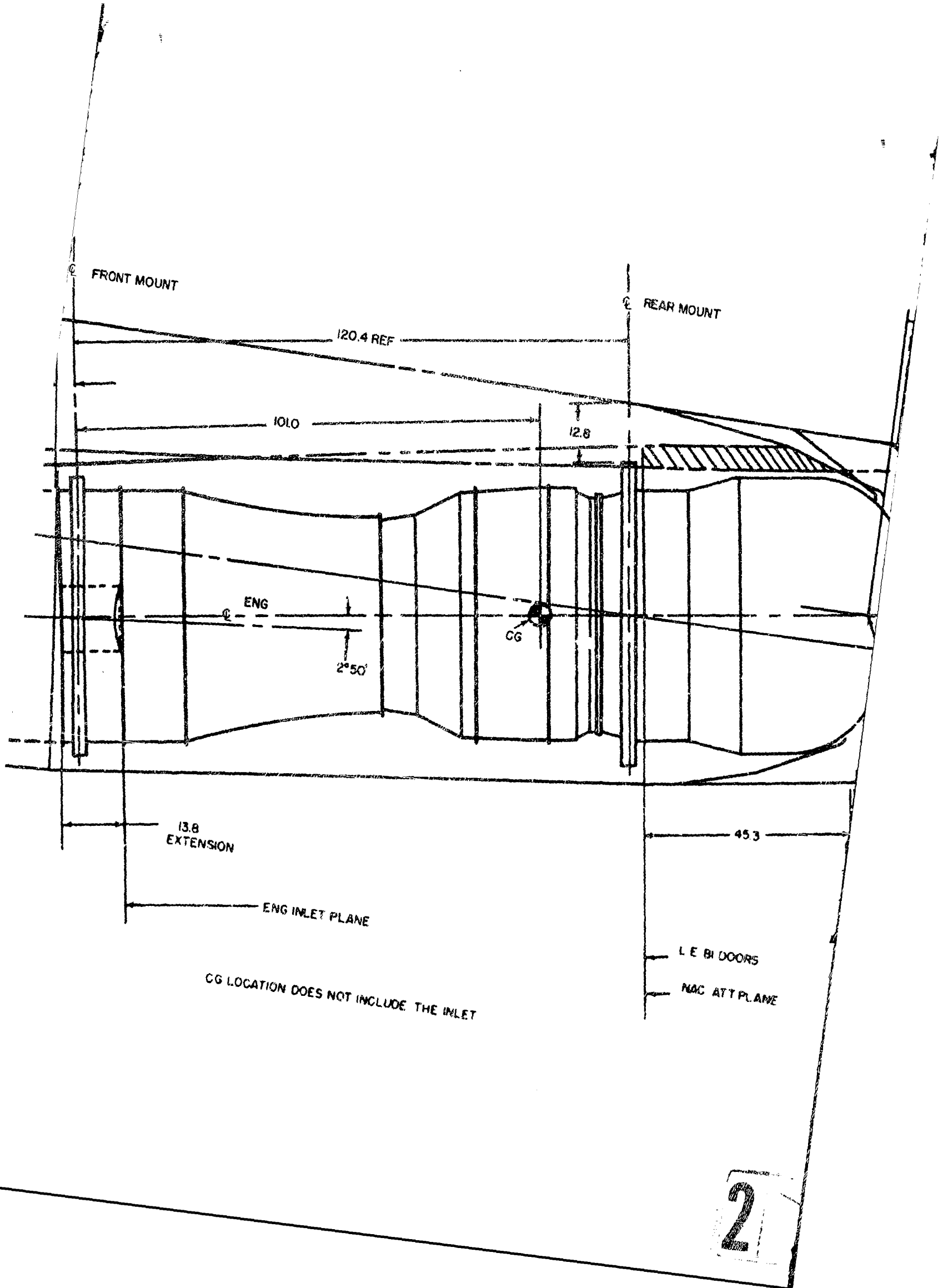
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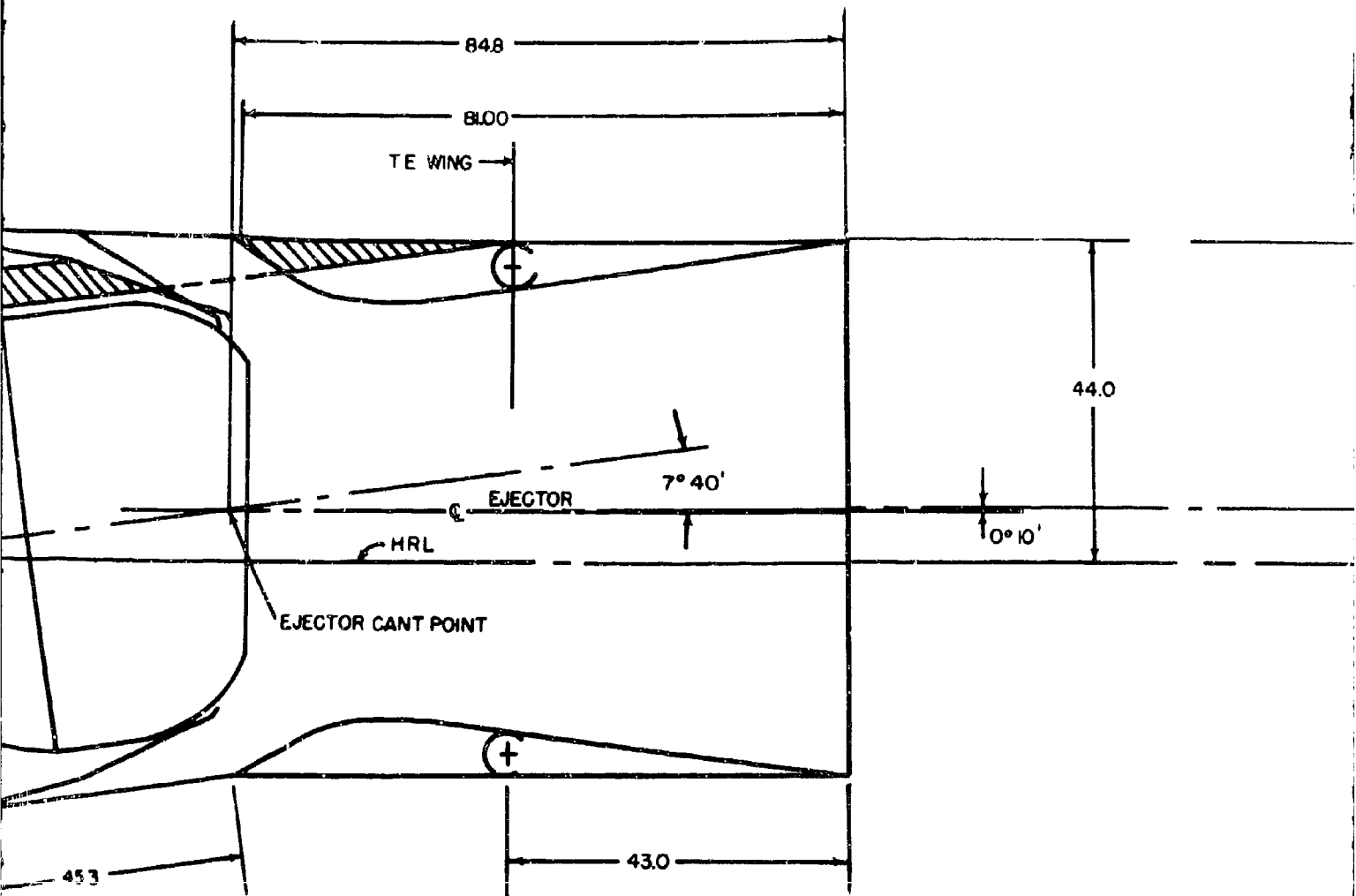
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PRATT & WHITNEY AIRCRAFT





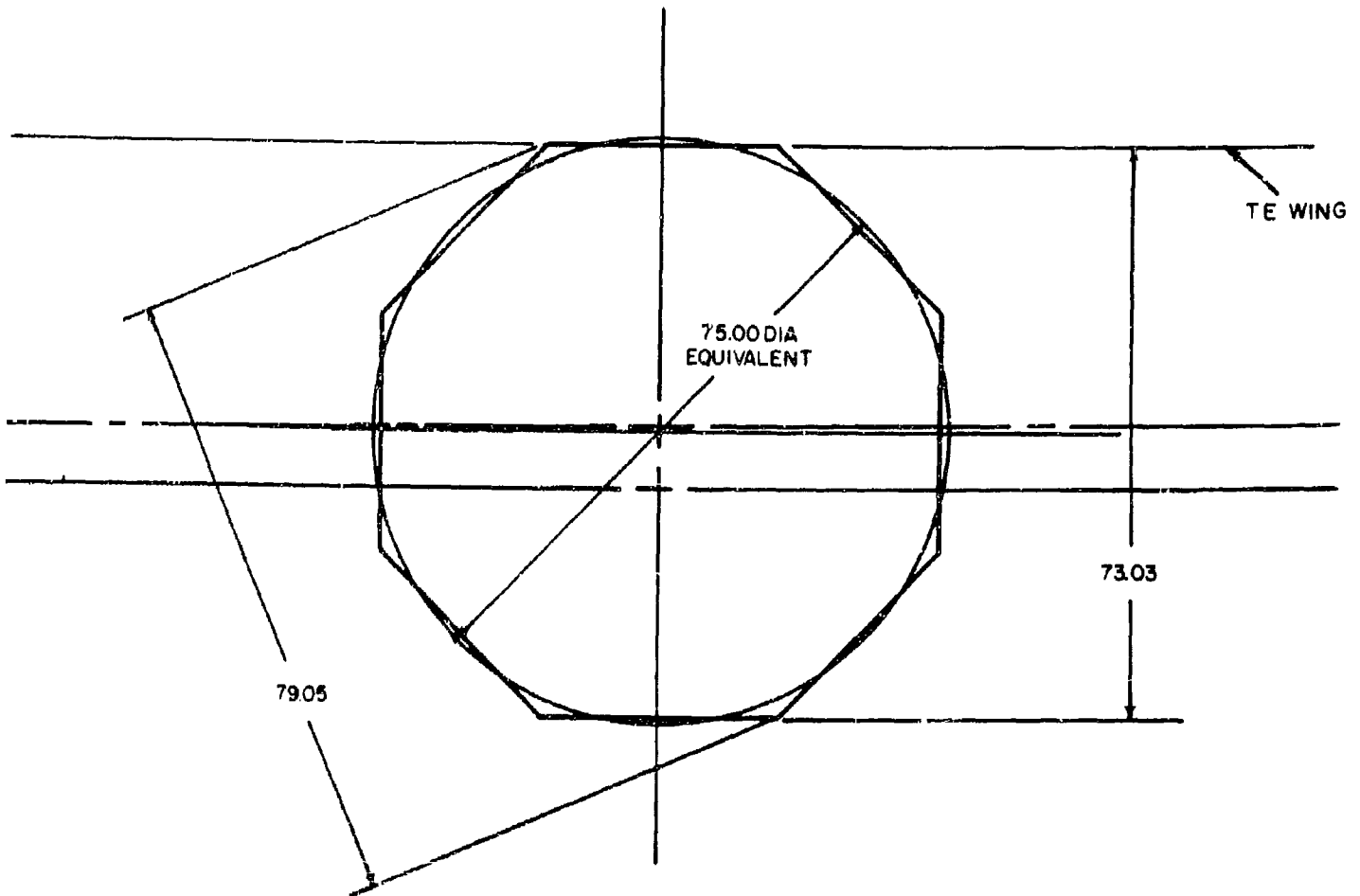


E IN DOORS

NAC ATT PLANE

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2300°F MAX. TURBINE INLET  
TEMPERATURE  
EJECTOR CANTED AT PRIMARY  
NOZZLE PLANE

STJ227 500 LBS./SEC. TURBOJET

Figure 1-16

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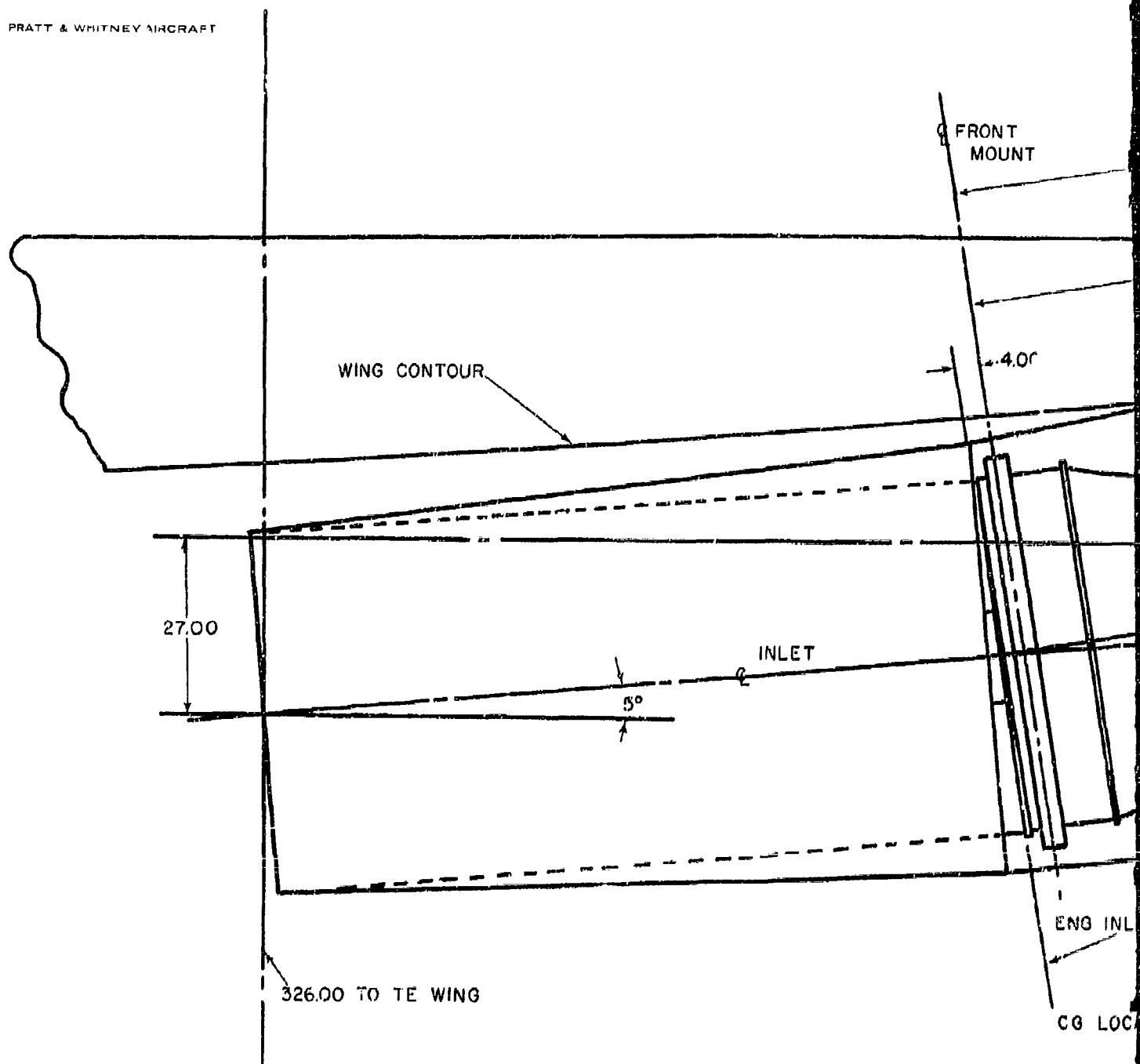
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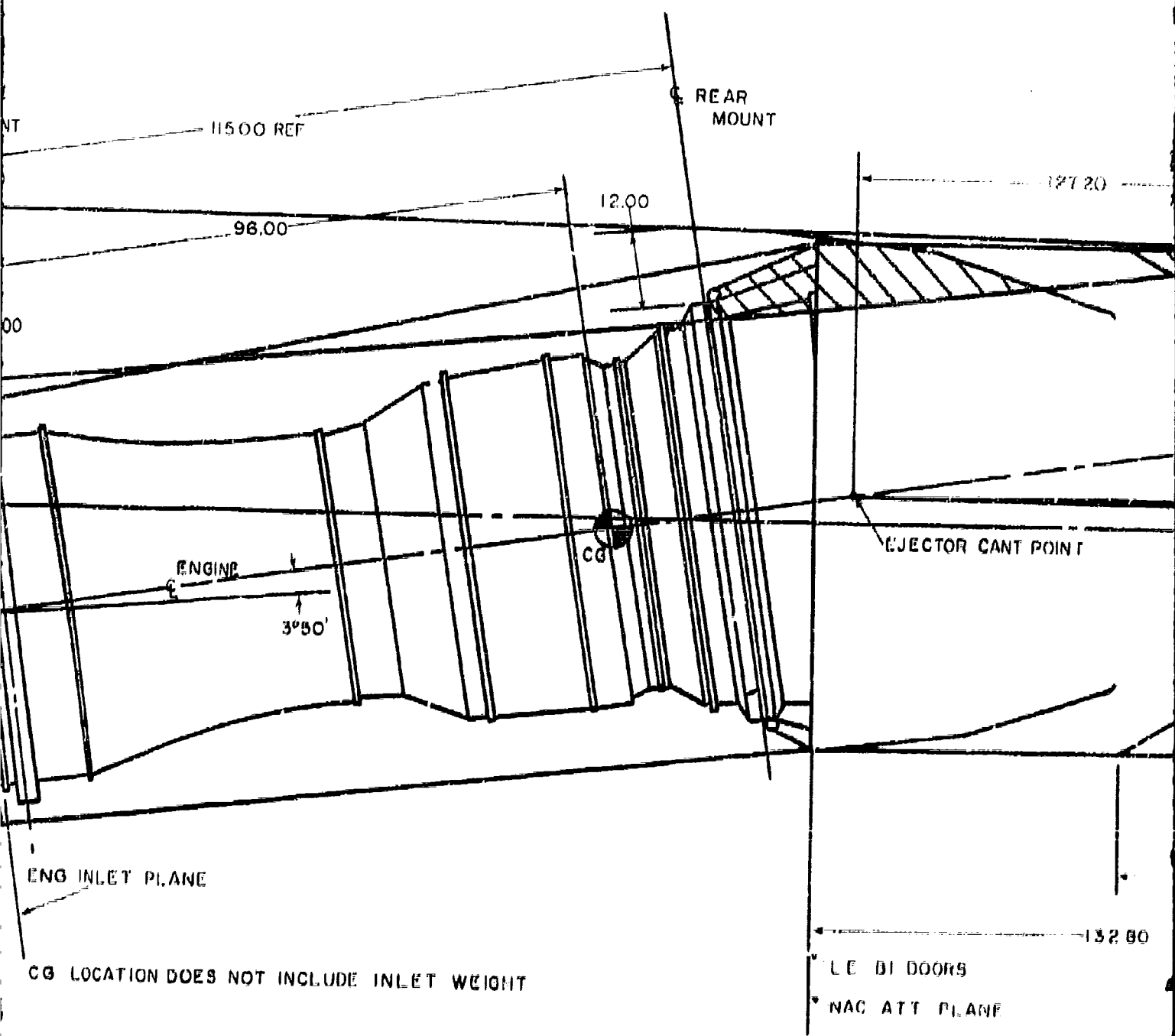
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SECTION 793 AND 794, THE TRANSMISSION OR THE  
REVELATION OF ITS CONTENTS IN ANY MANNER TO  
AN UNAUTHORIZED PERSON IS PROHIBITED BY LAW

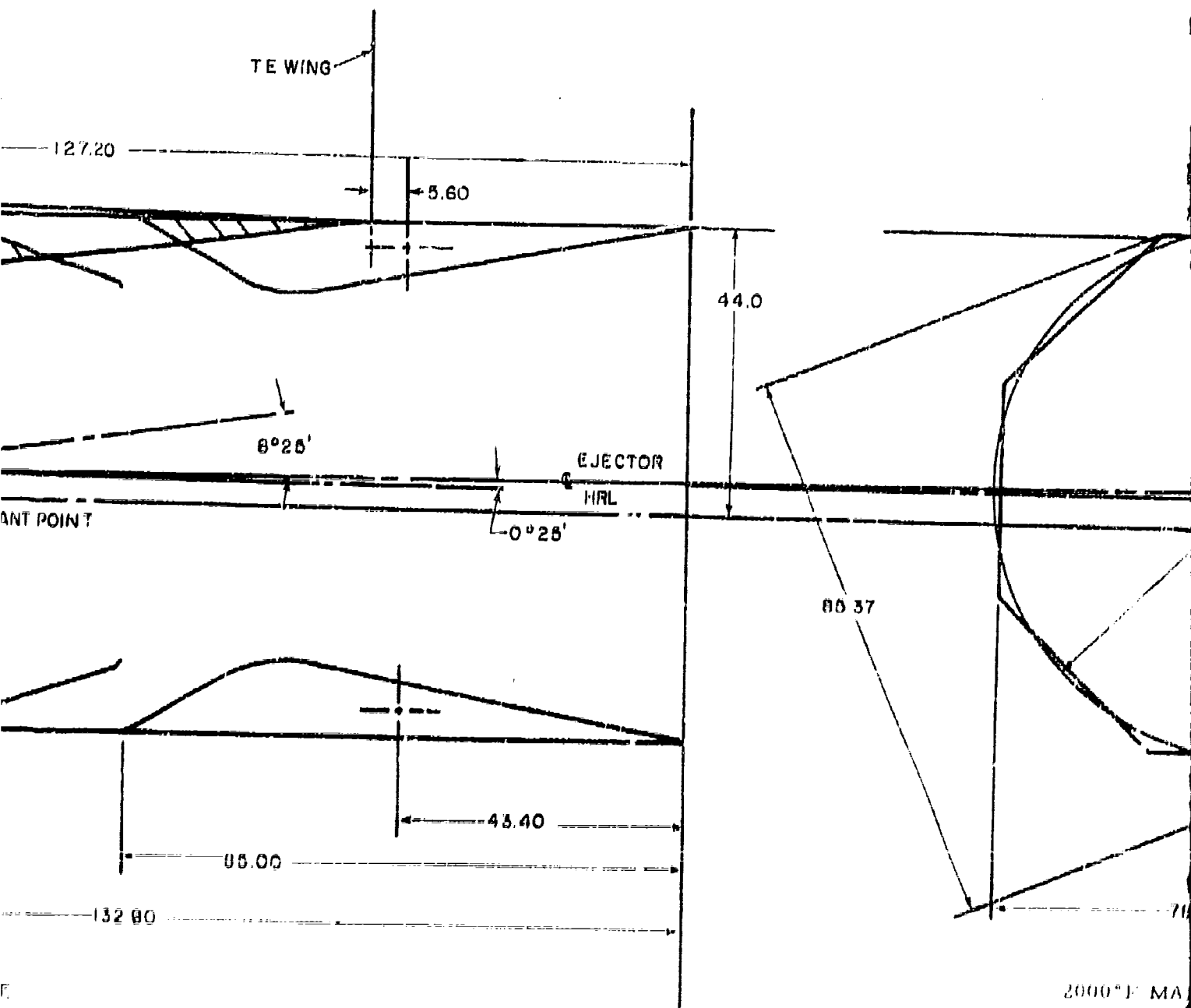
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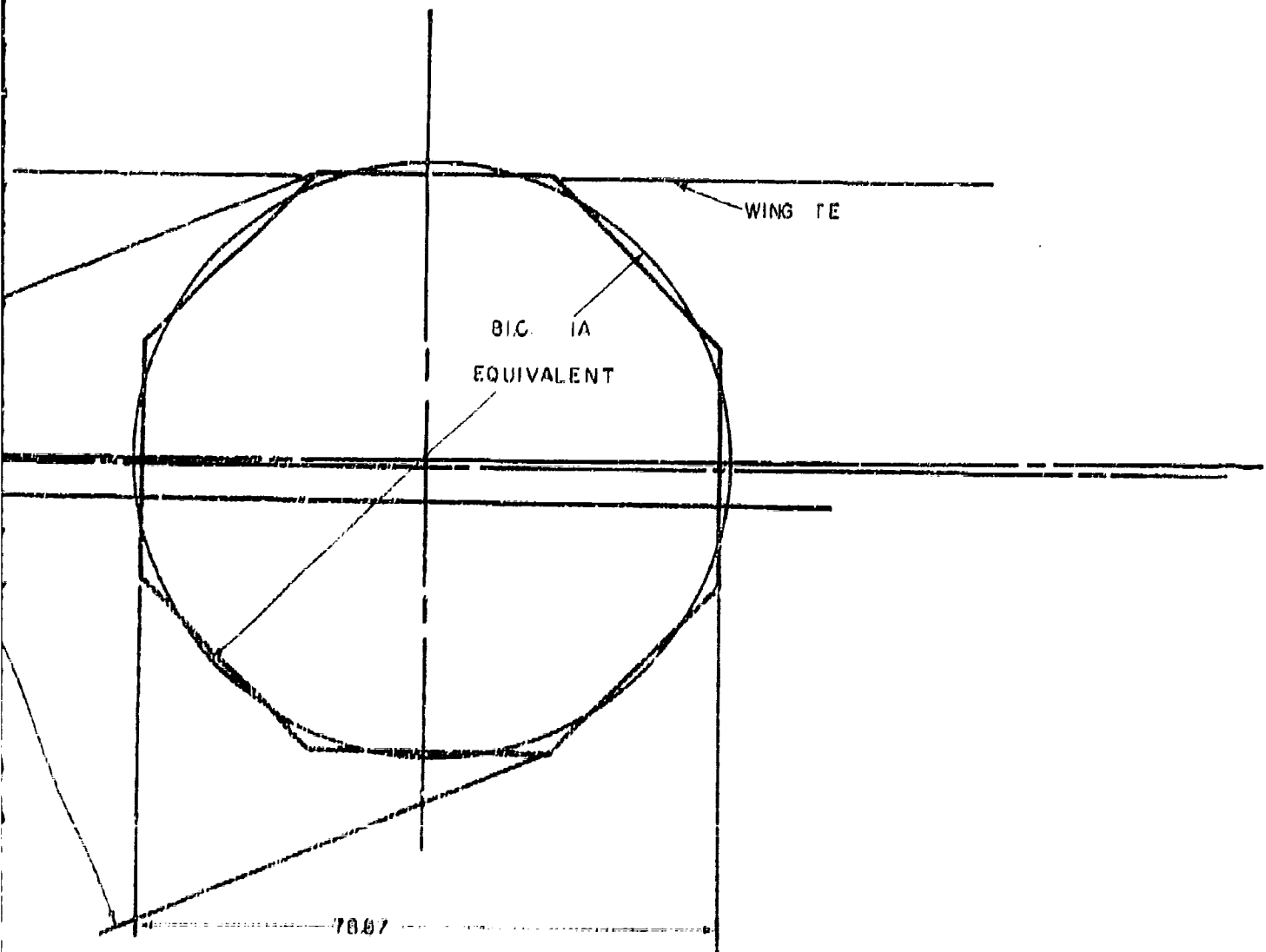




2000°F MAX  
TEMPERATURE  
EJECTOR  
MOUNTING

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2000°F MAX. TURBINE INLET  
TEMPERATURE  
EJECTOR CANTED AT REAR  
MOUNT PLANE

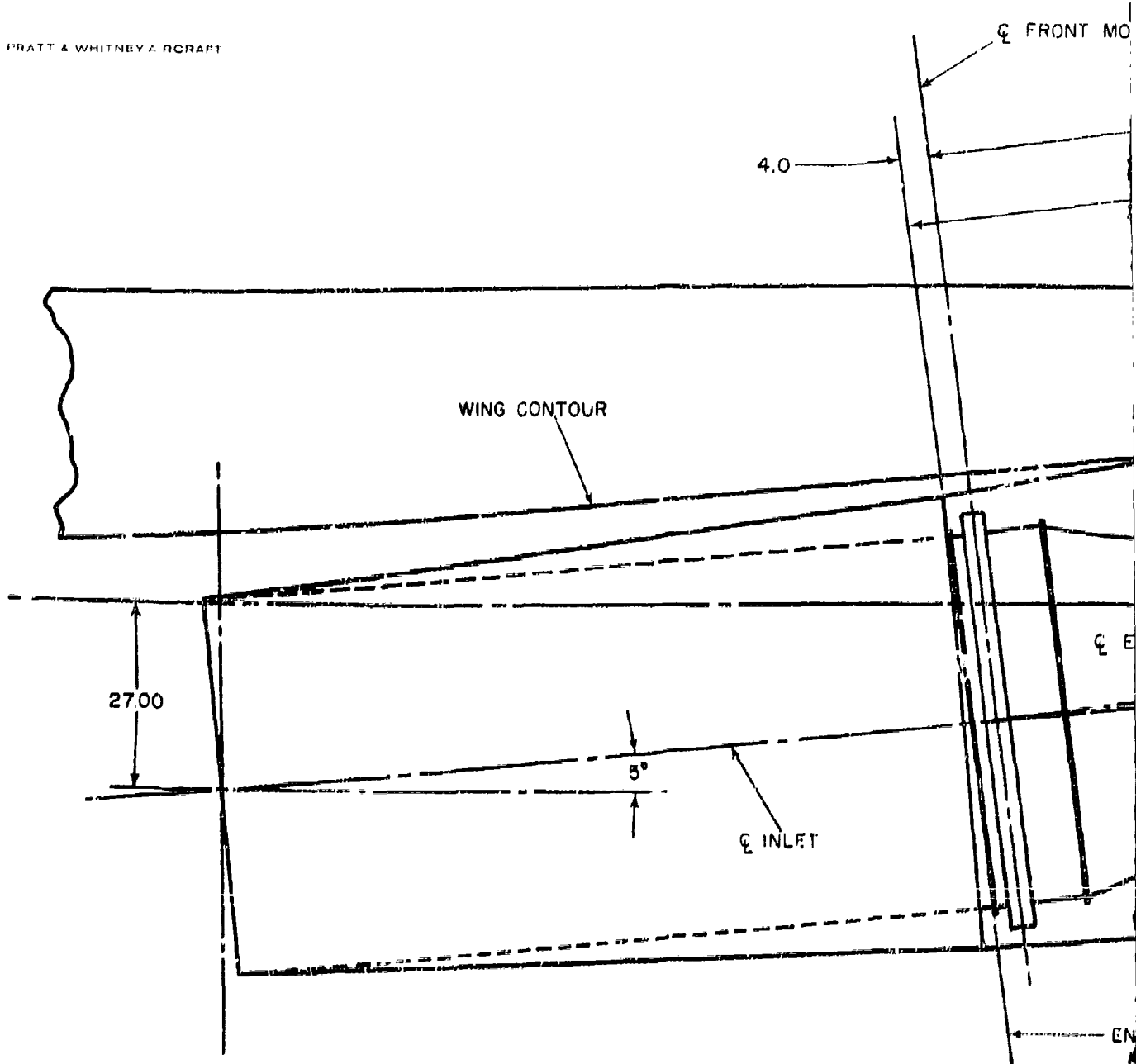
STJ227 500 LBS./SEC. TURBOJET

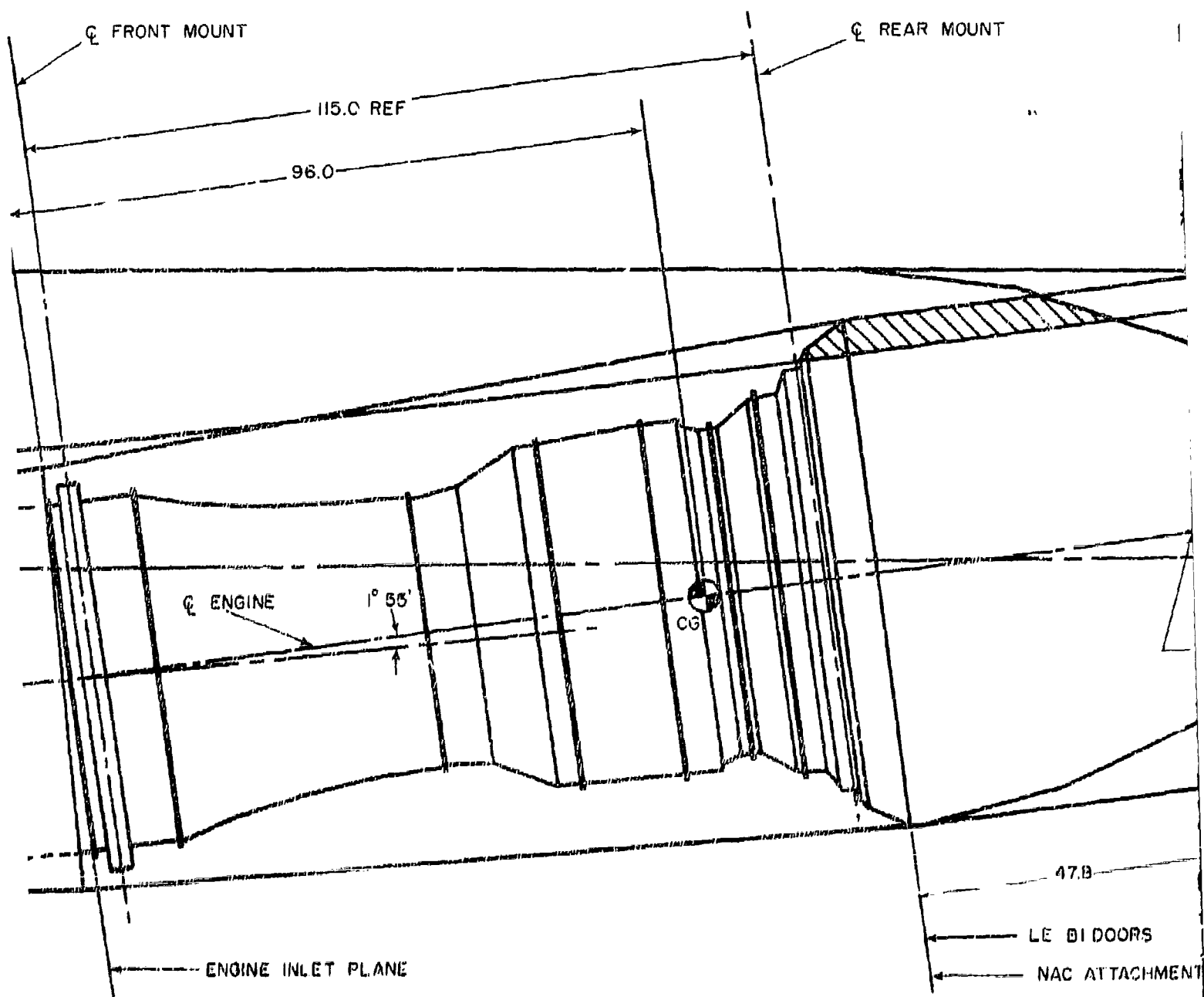
Figure 1-17

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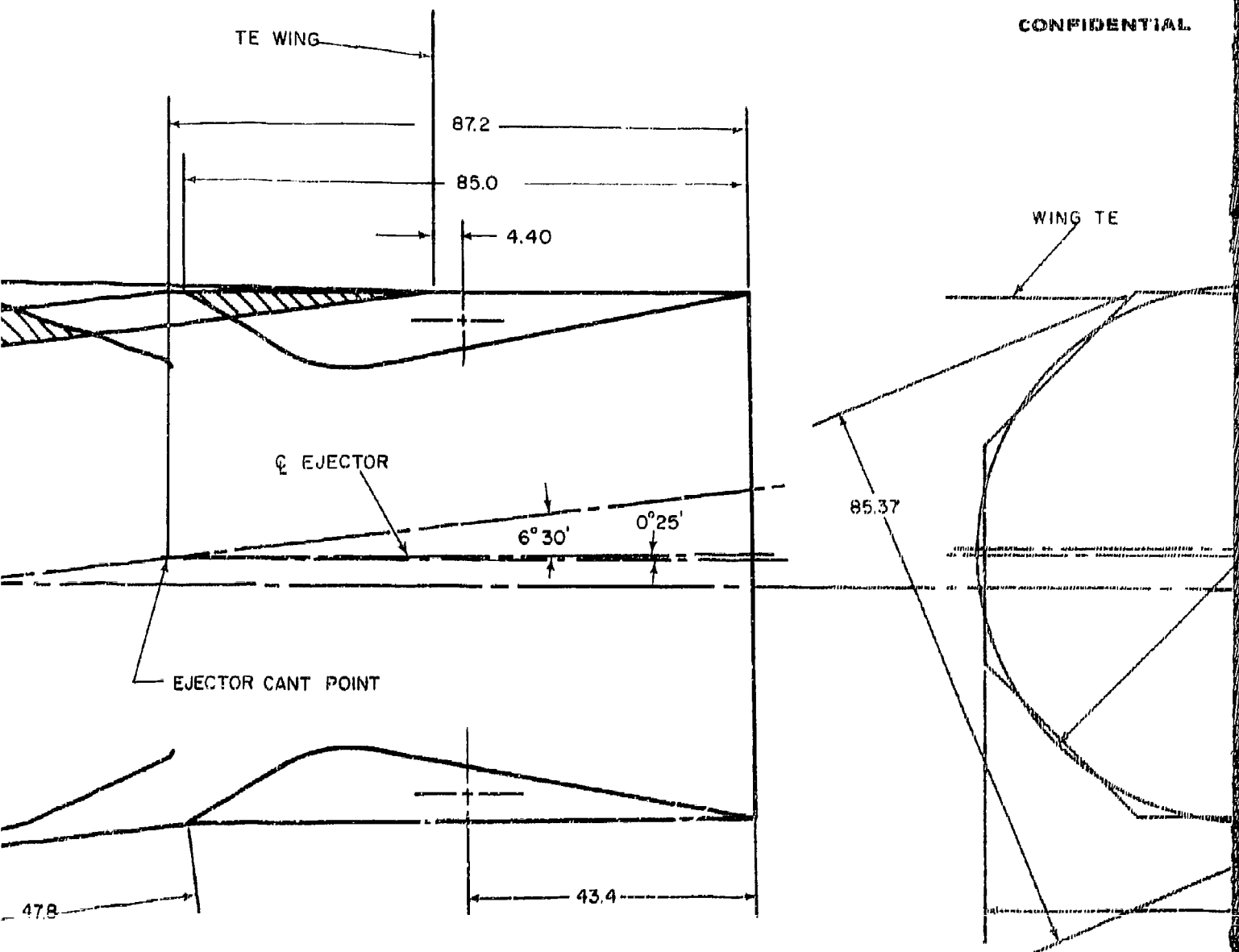
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E 81 DOORS

NAC ATTACHMENT PLANE

CG LOCATION DOES NOT INCLUDE INLET WEIGHT

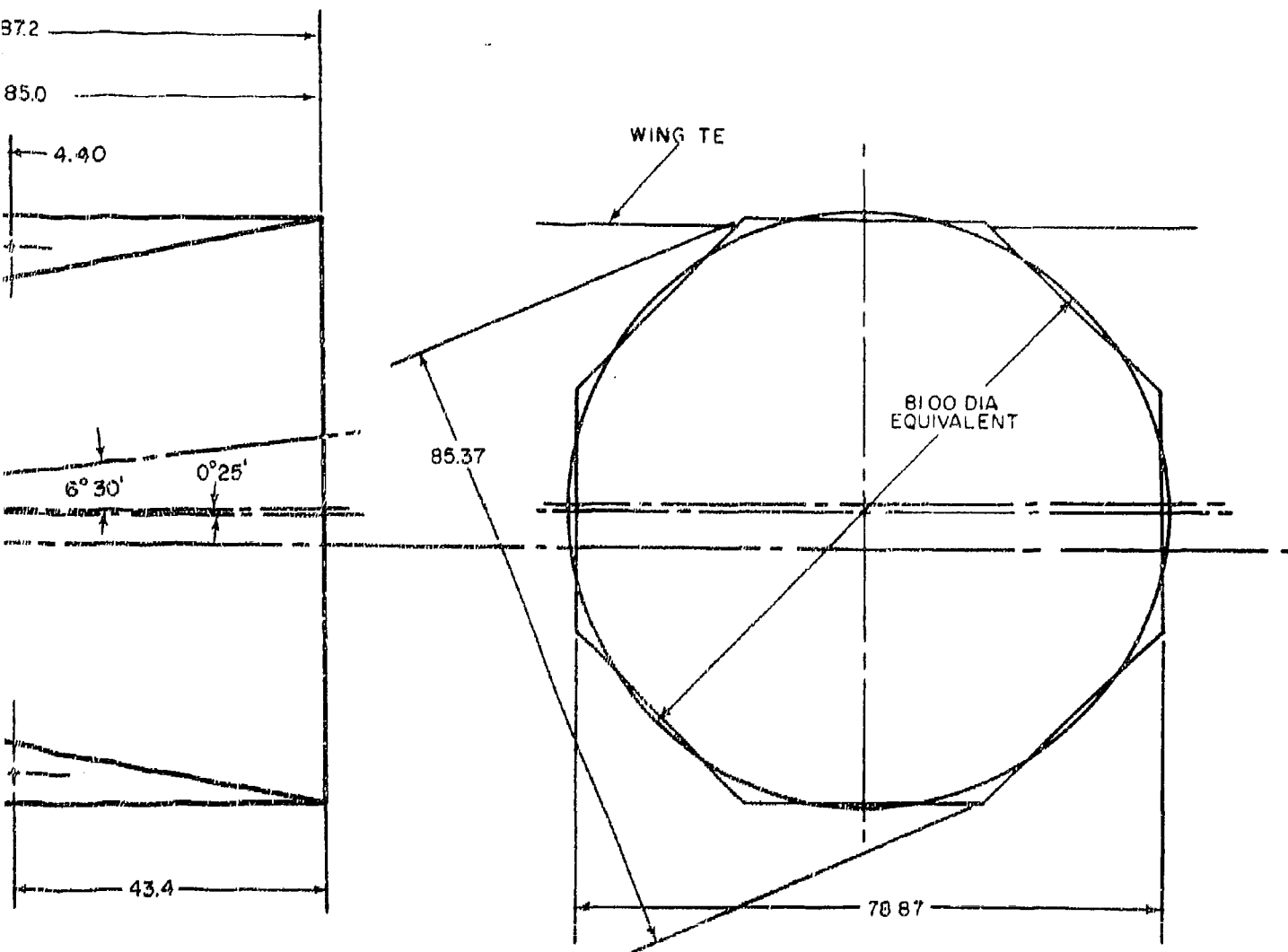
2000°F MAX TU  
TEMPERATUR  
EJECTOR CANT  
NOZZLE PLAN

STJ227 500 LBS./SEC. TU

Figure 1-18

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CG LOCATION DOES NOT INCLUDE INLET WEIGHT

2000°F MAX. TURBINE INLET  
TEMPERATURE  
EJECTOR CANTED AT PRIMARY  
NOZZLE PLANE

STJ227 500 LBS./SEC. TURBOJET

Figure 1-18

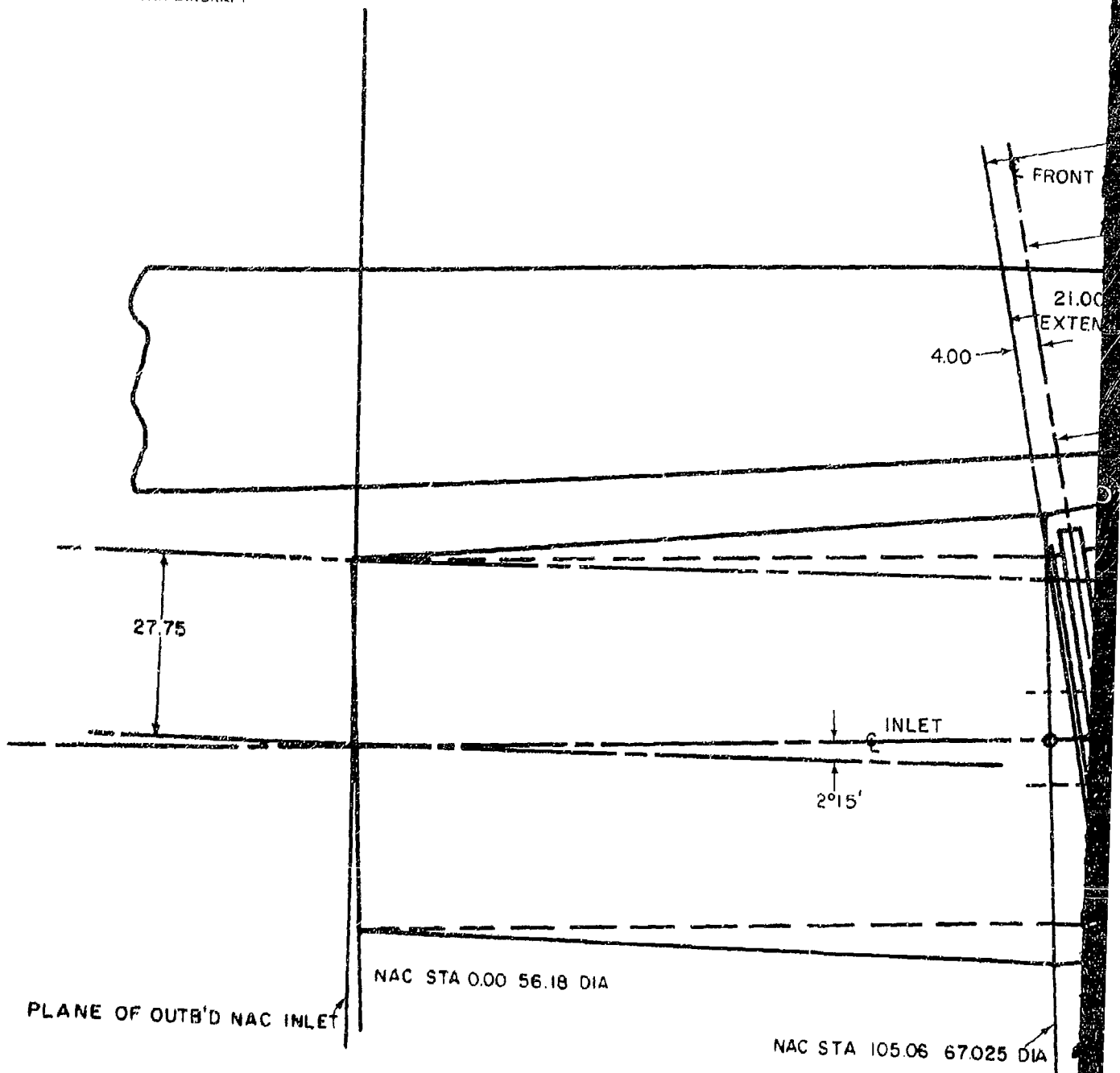
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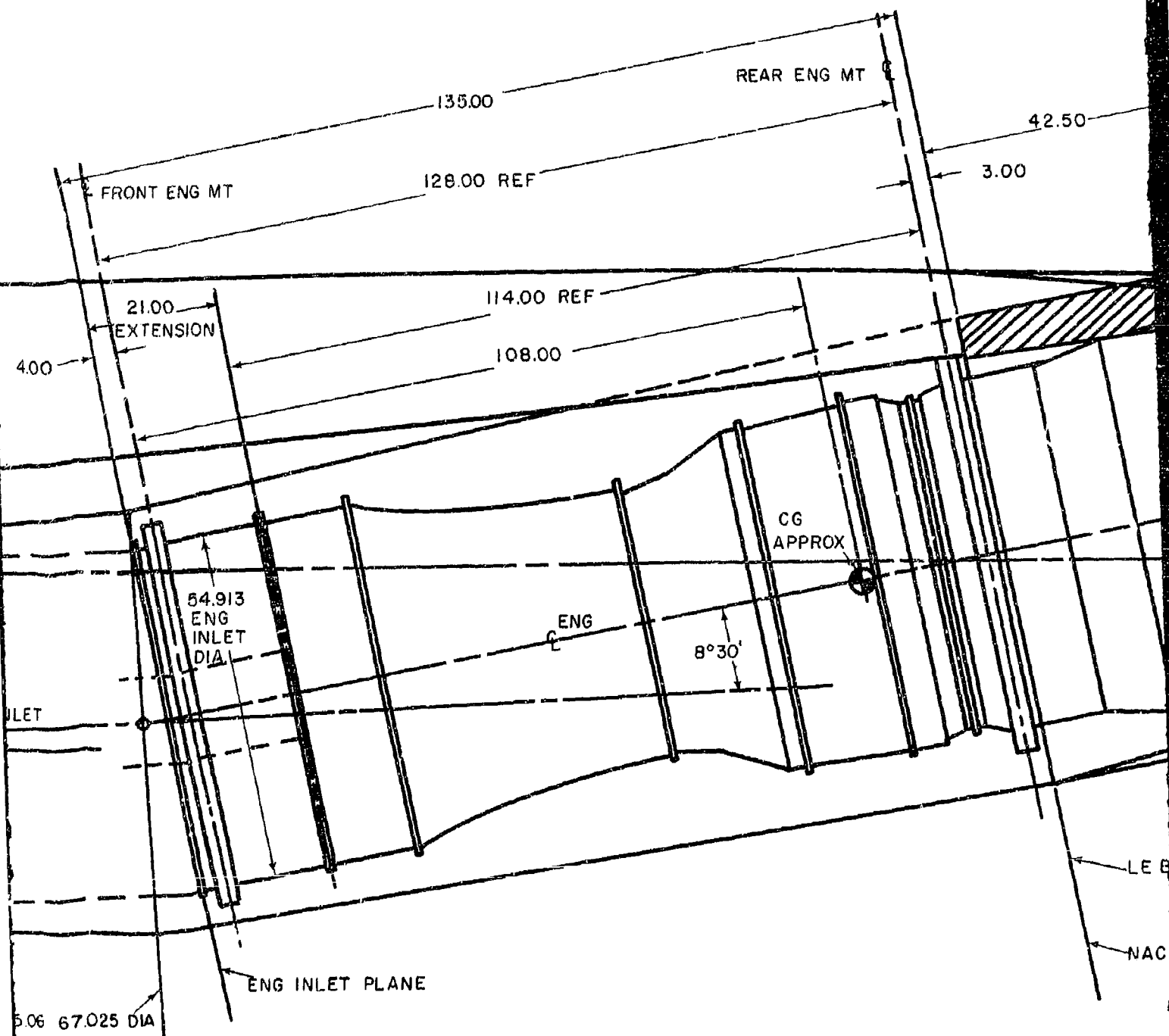
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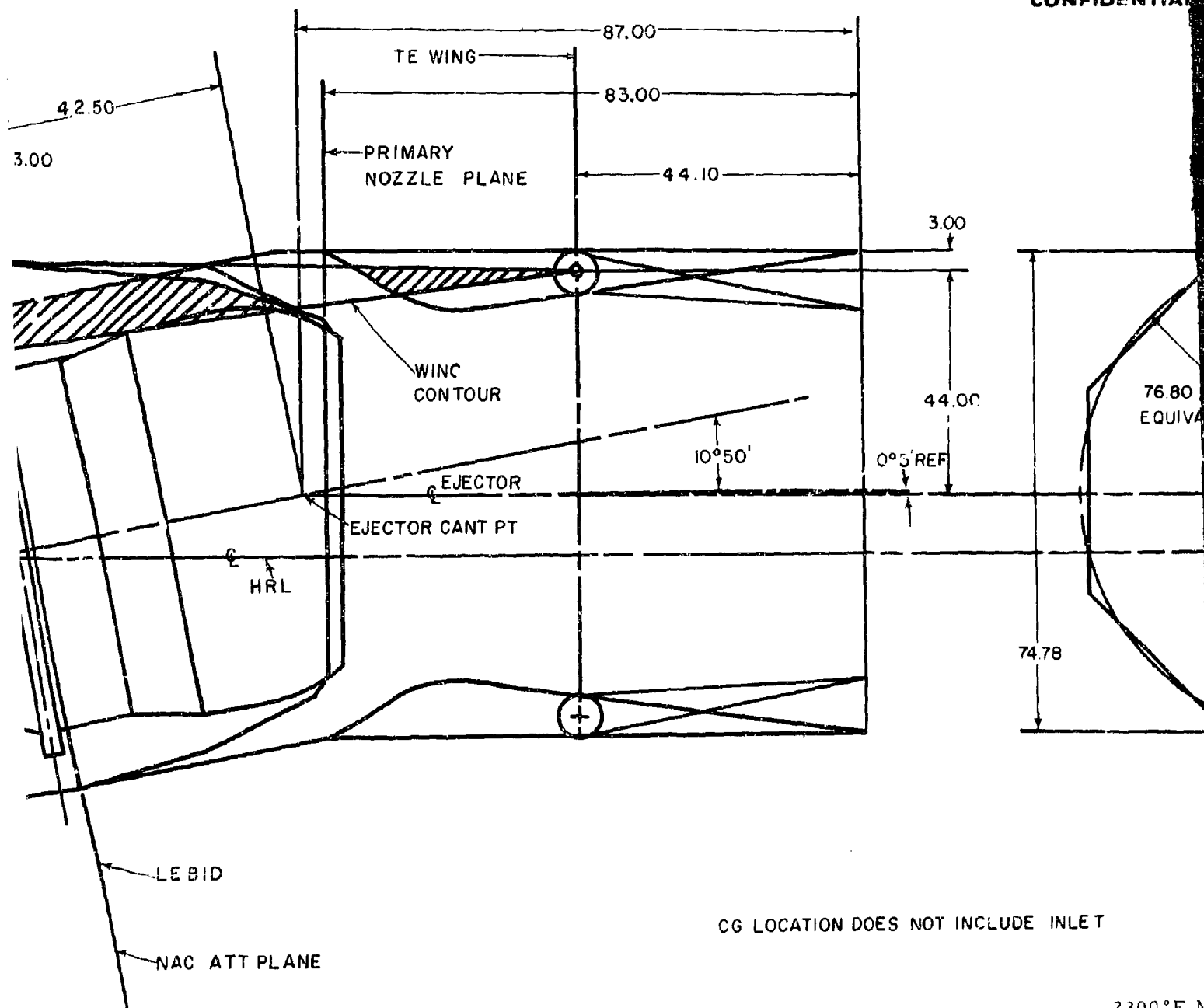


PRATT & WHITNEY AIRCRAFT





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CG LOCATION DOES NOT INCLUDE INLET

2300°F M  
TEMPE  
EJECTOR  
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\*NOTE:

STJ227 525 LBS./SEC.

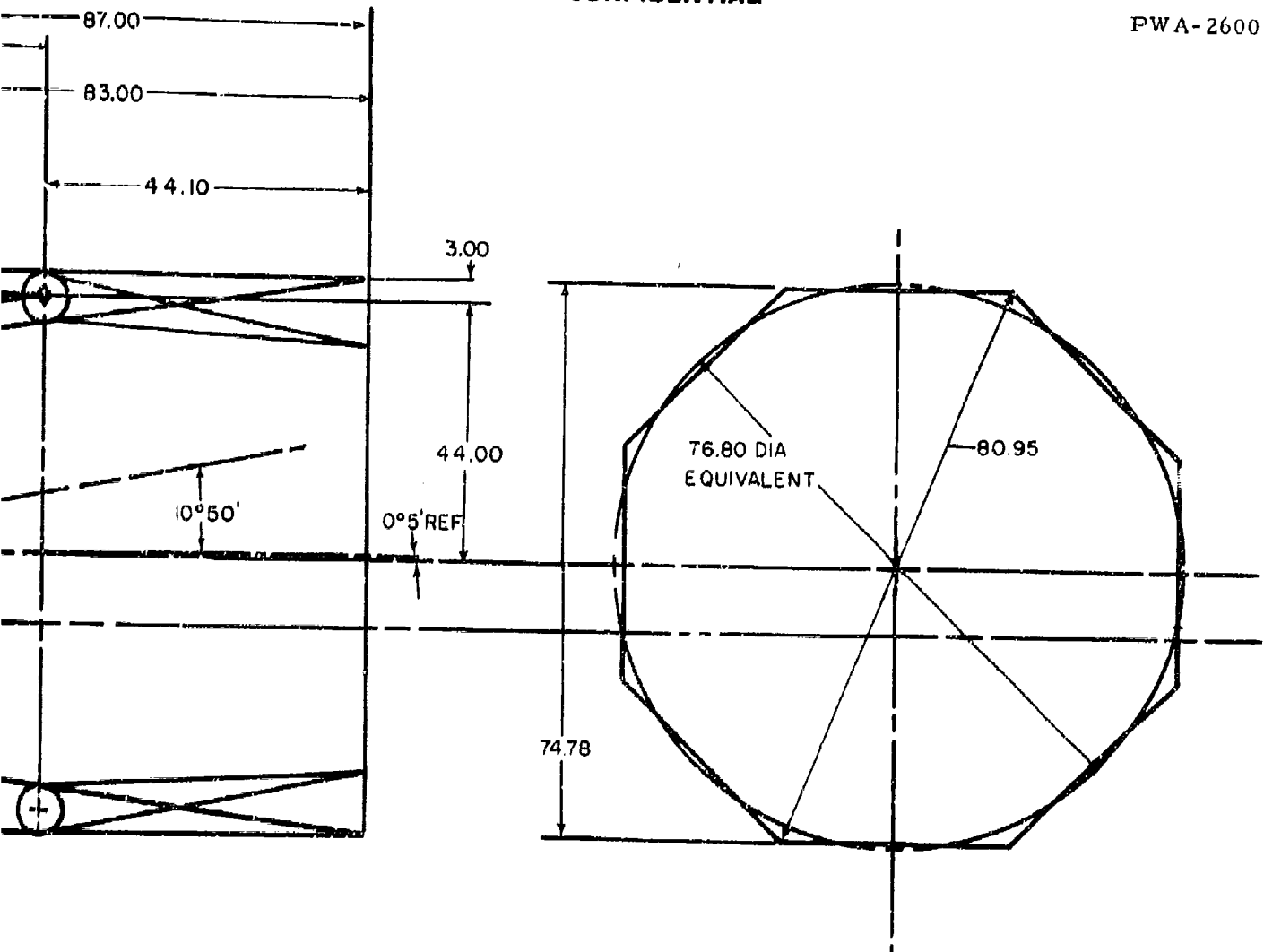
Figure 1-19

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3

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CG LOCATION DOES NOT INCLUDE INLET

2300°F MAXIMUM TURBINE INLET  
TEMPERATURE

EJECTOR CANTED 4.00 FWD. OF  
PRIMARY NOZZLE PLANE

\*NOTE: CG LOCATION DOES NOT  
INCLUDE INLET

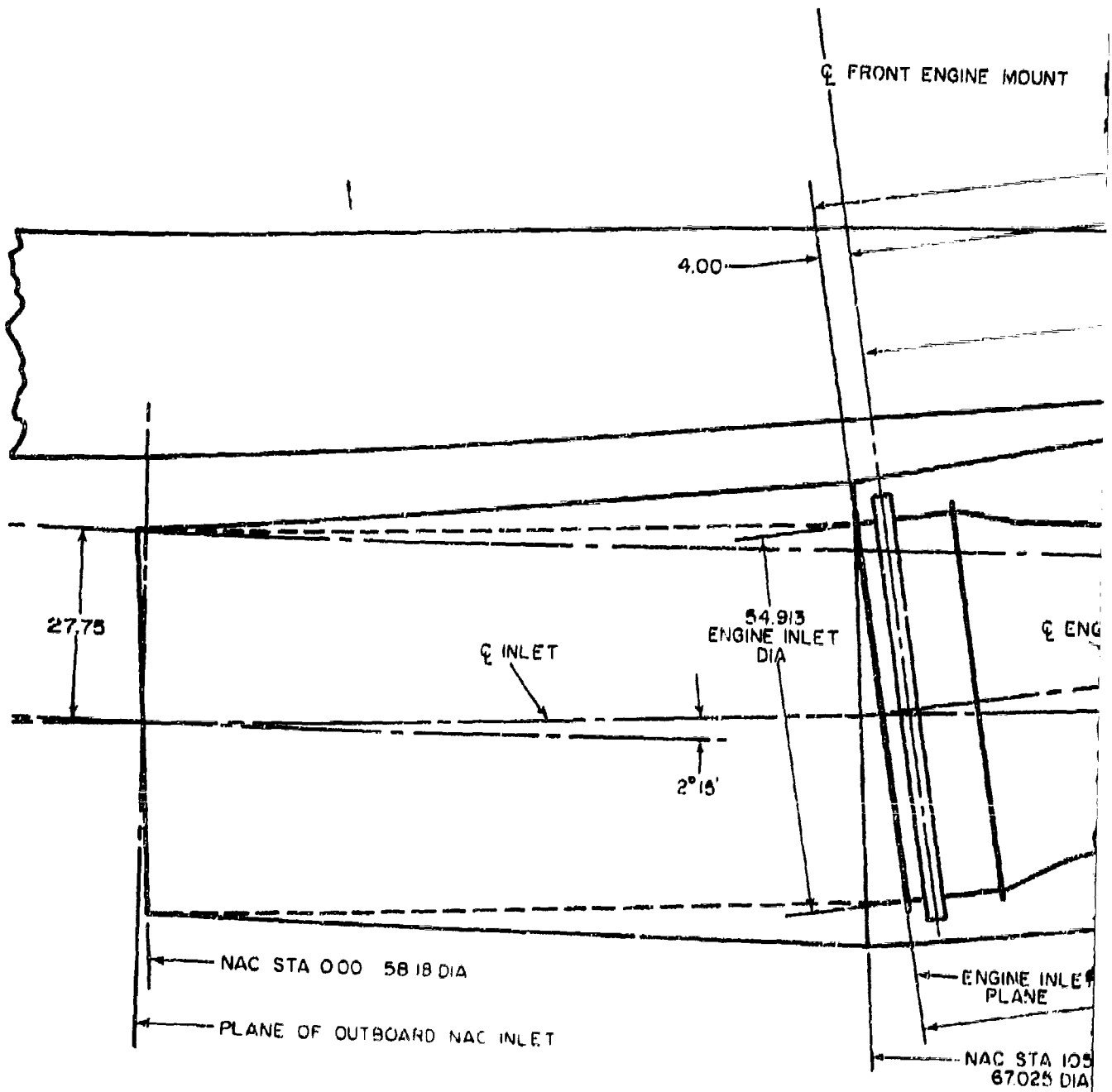
STJ227 525 LBS./SEC. TURBOJET

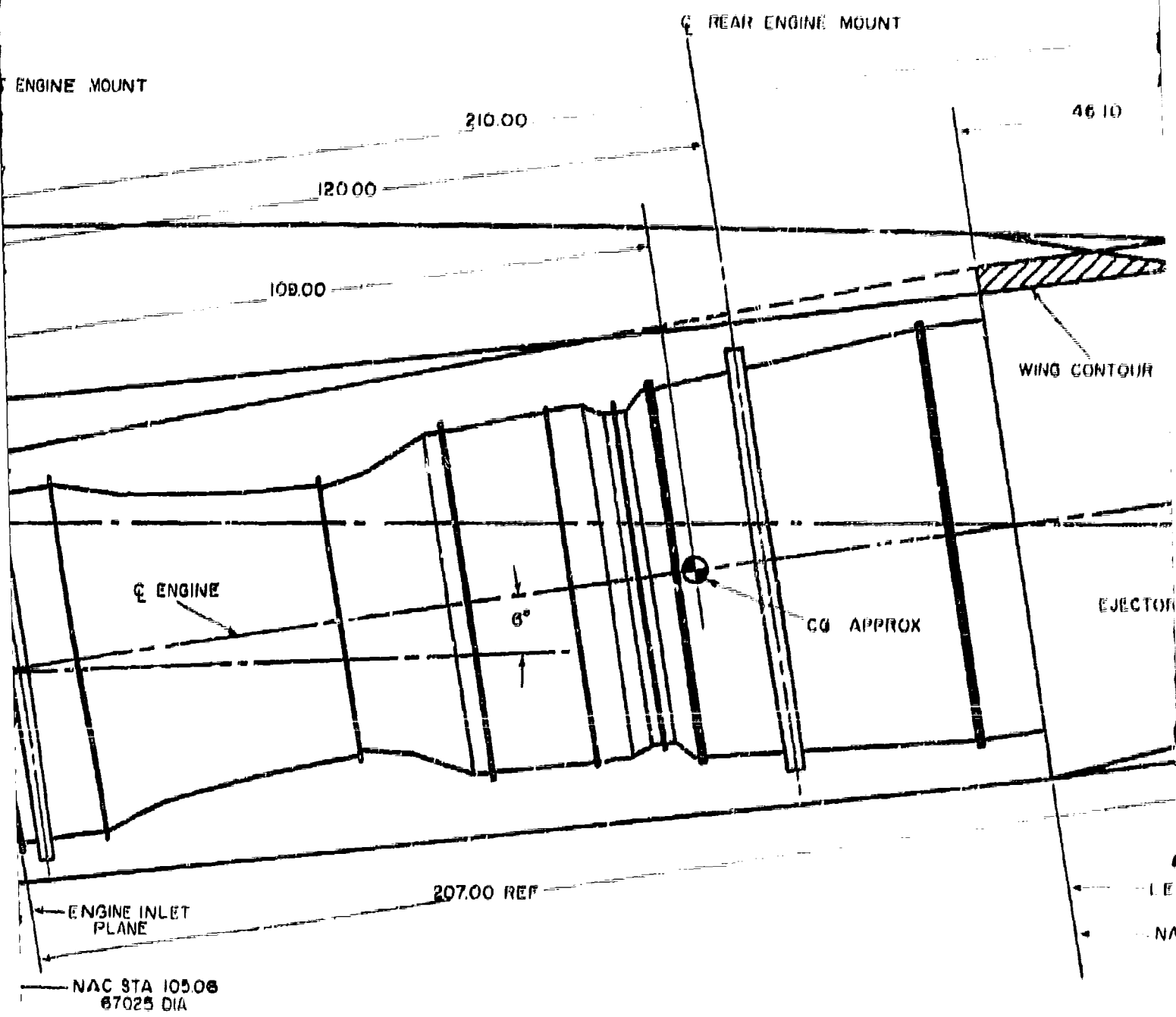
Figure 1-19

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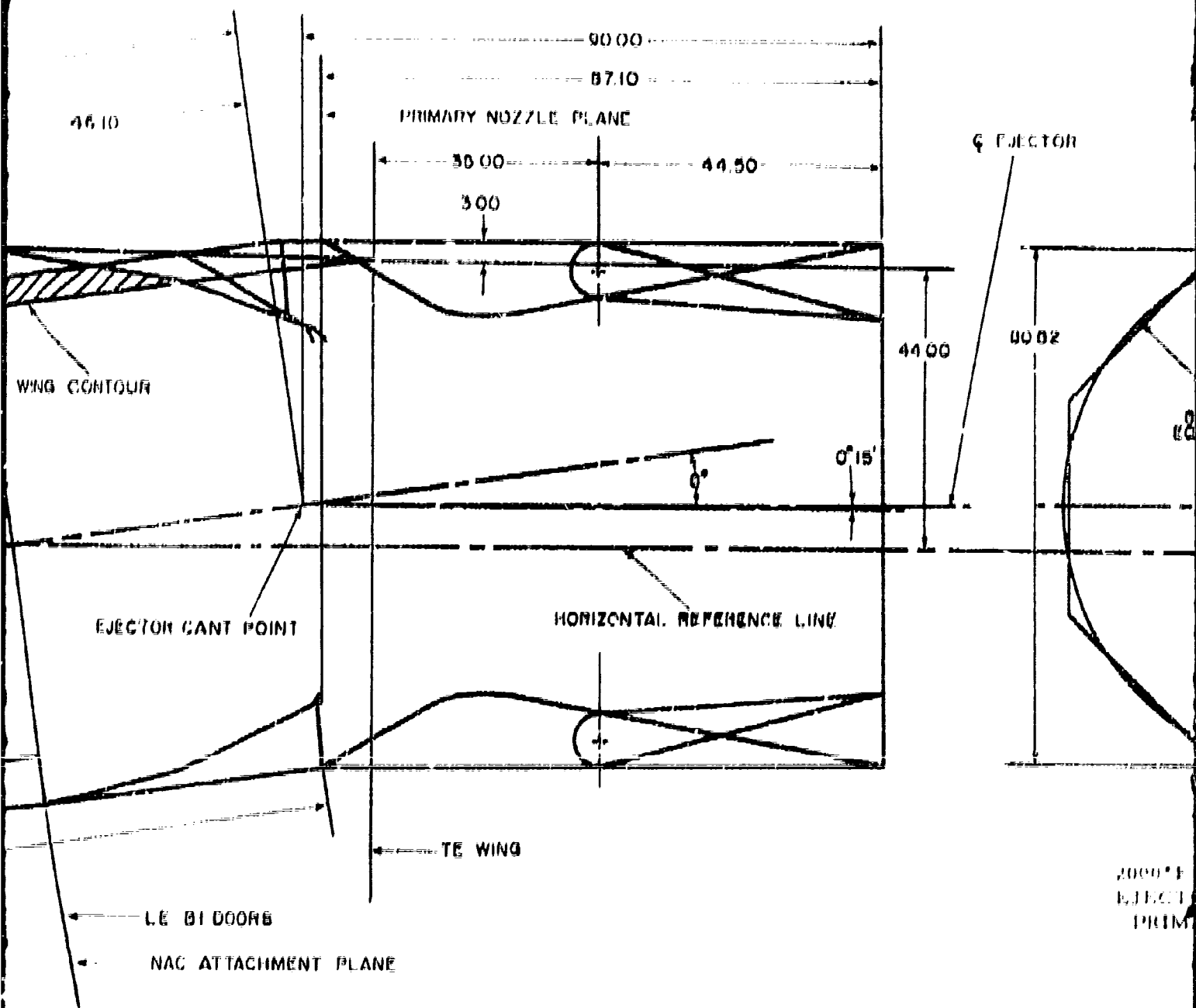
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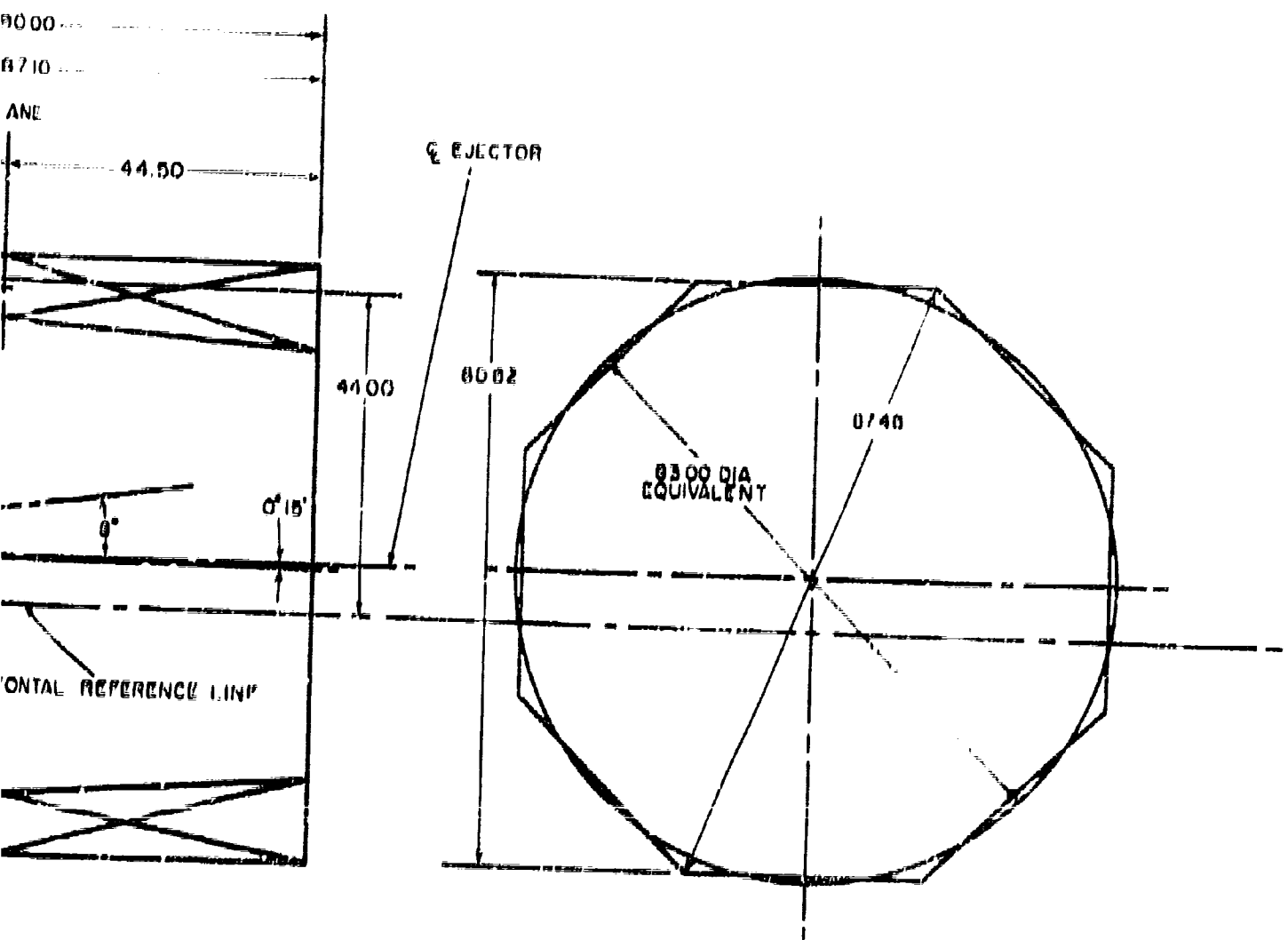


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2000°F MAXIMUM TURBINE INLET  
EJECTOR CANTED 2° 30' FWD OF  
PRIMARY NOZZLE PLANE

5TJ227 525 LBS./SEC. TURBOJET

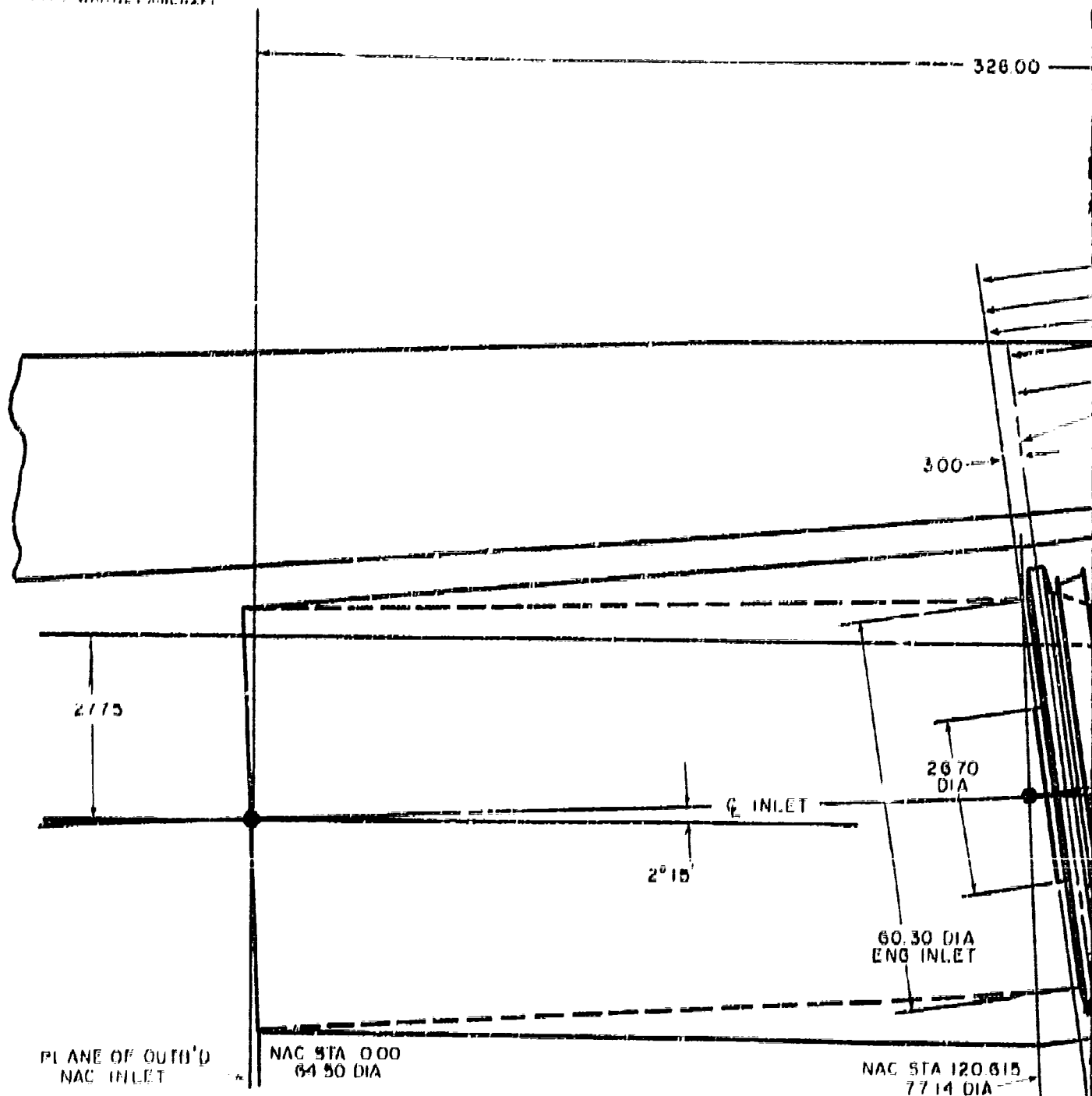
Figure 1-20

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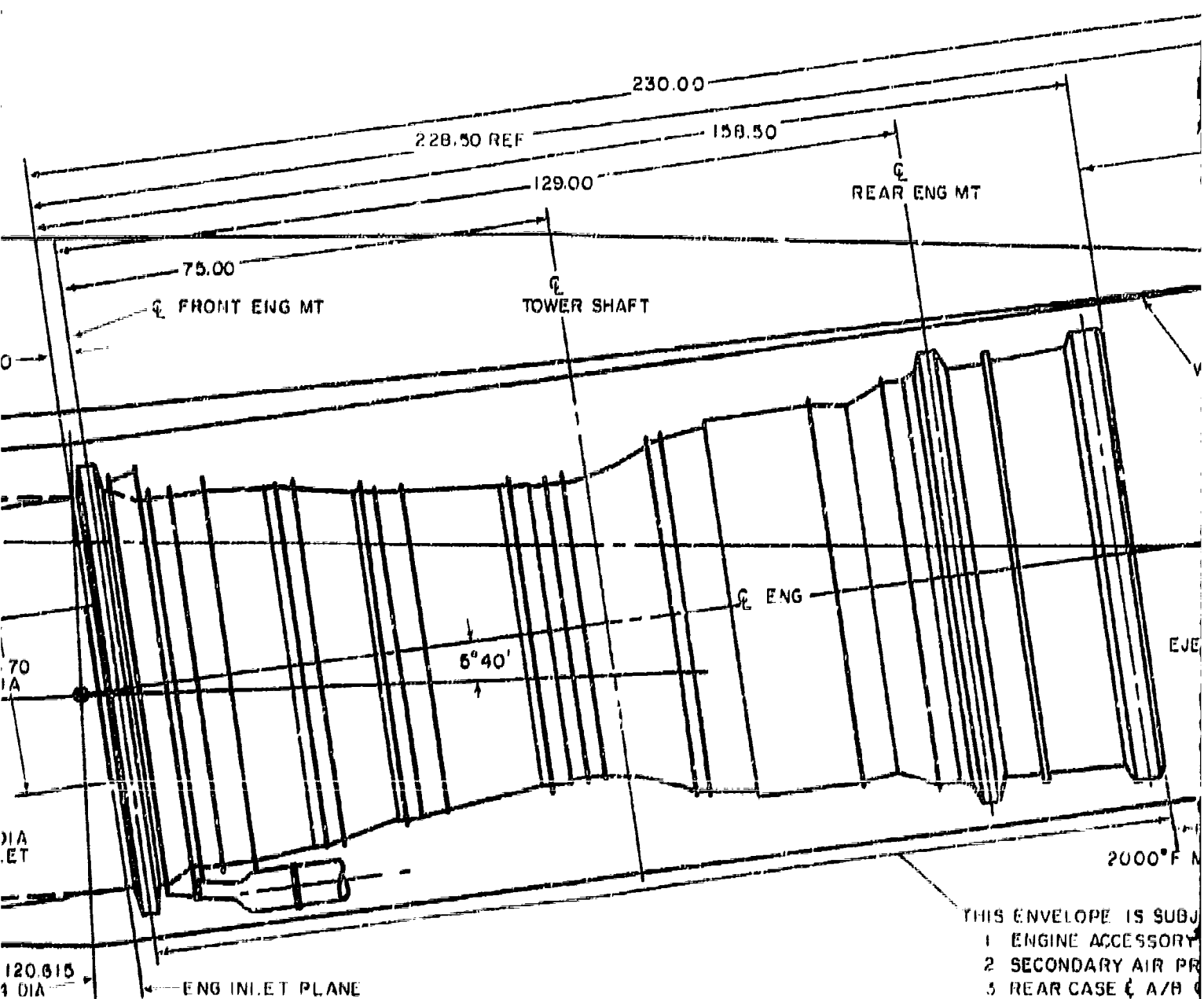
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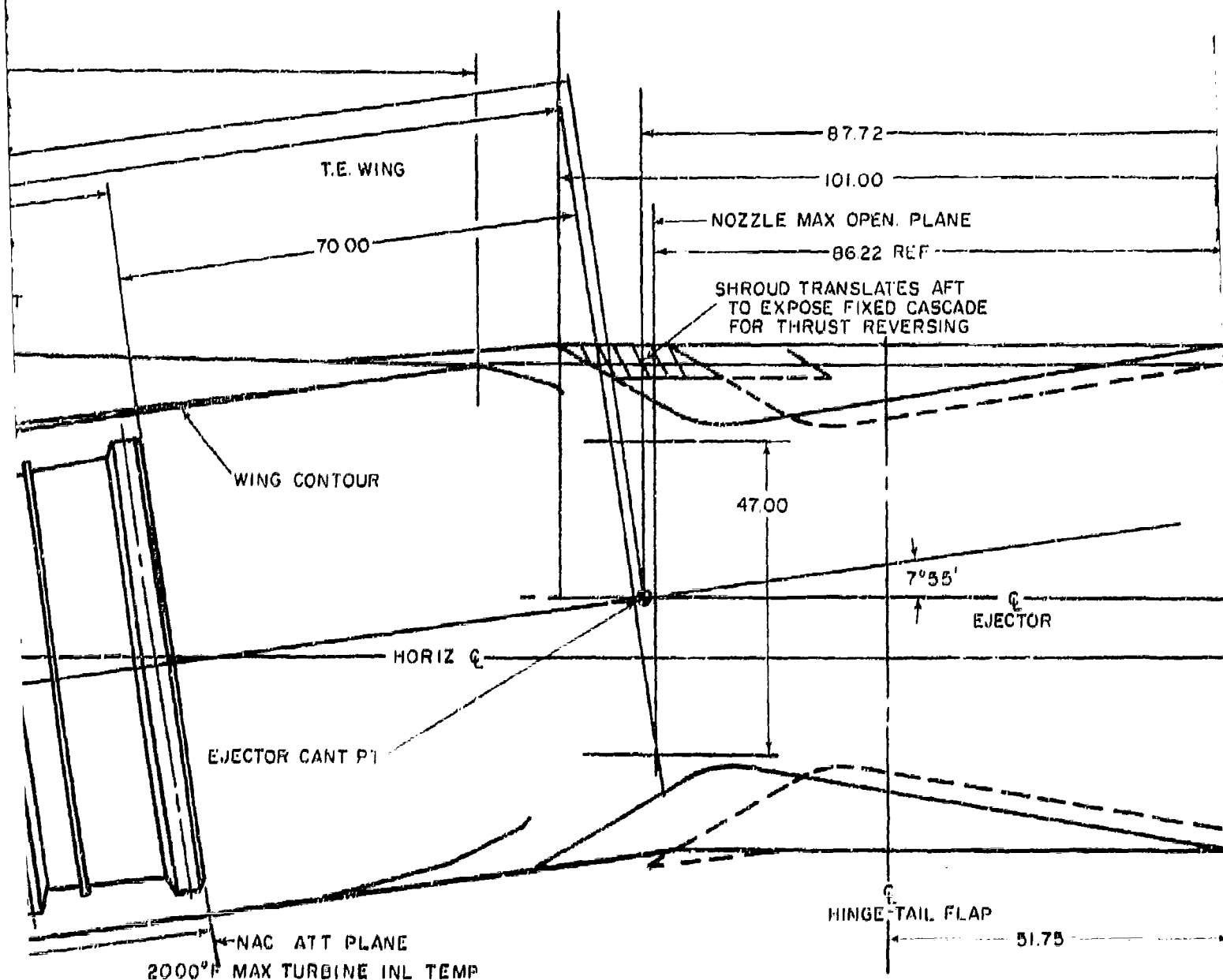


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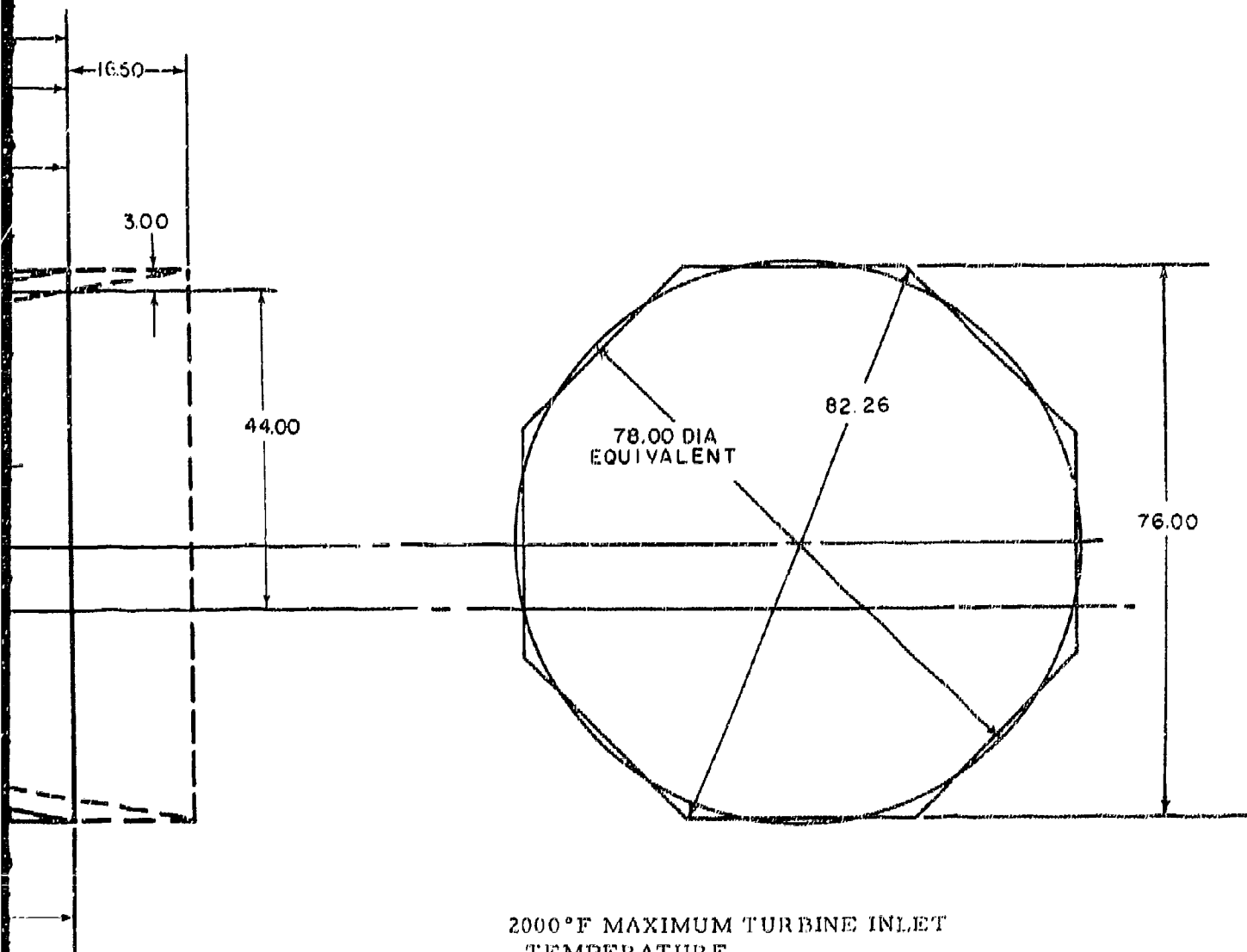




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 2. ENGINE ACCESSORY ARRANGEMENT  
 3. SECONDARY AIR PROVISIONS FOR EJECTOR  
 4. REAR CASE & A/O COOLING PROVISIONS

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2000°F MAXIMUM TURBINE INLET  
TEMPERATURE  
EJECTOR CANTED 1.50 FWD. OF  
PRIMARY NOZZLE PLANE

THIS ENVELOPE IS SUBJECT TO CHANGE FOR:

1. ENGINE ACCESSORY ARRANGEMENT
2. SECONDARY AIR PROVISIONS FOR EJECTOR
3. REAR CASE & A/B COOLING PROVISIONS

STJ227, 525 LBS./SEC. (HIGH FLOW) TURBOJET

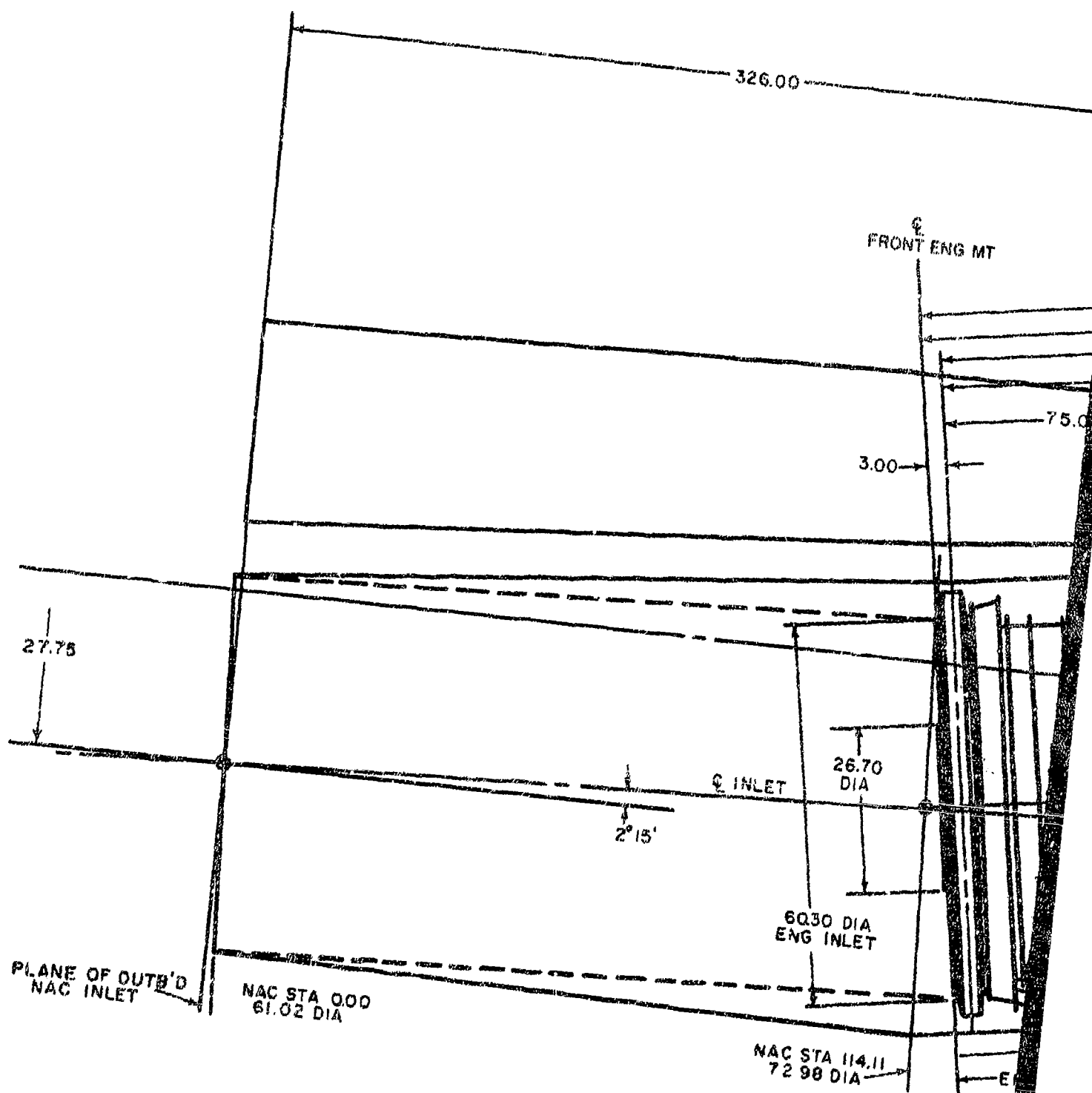
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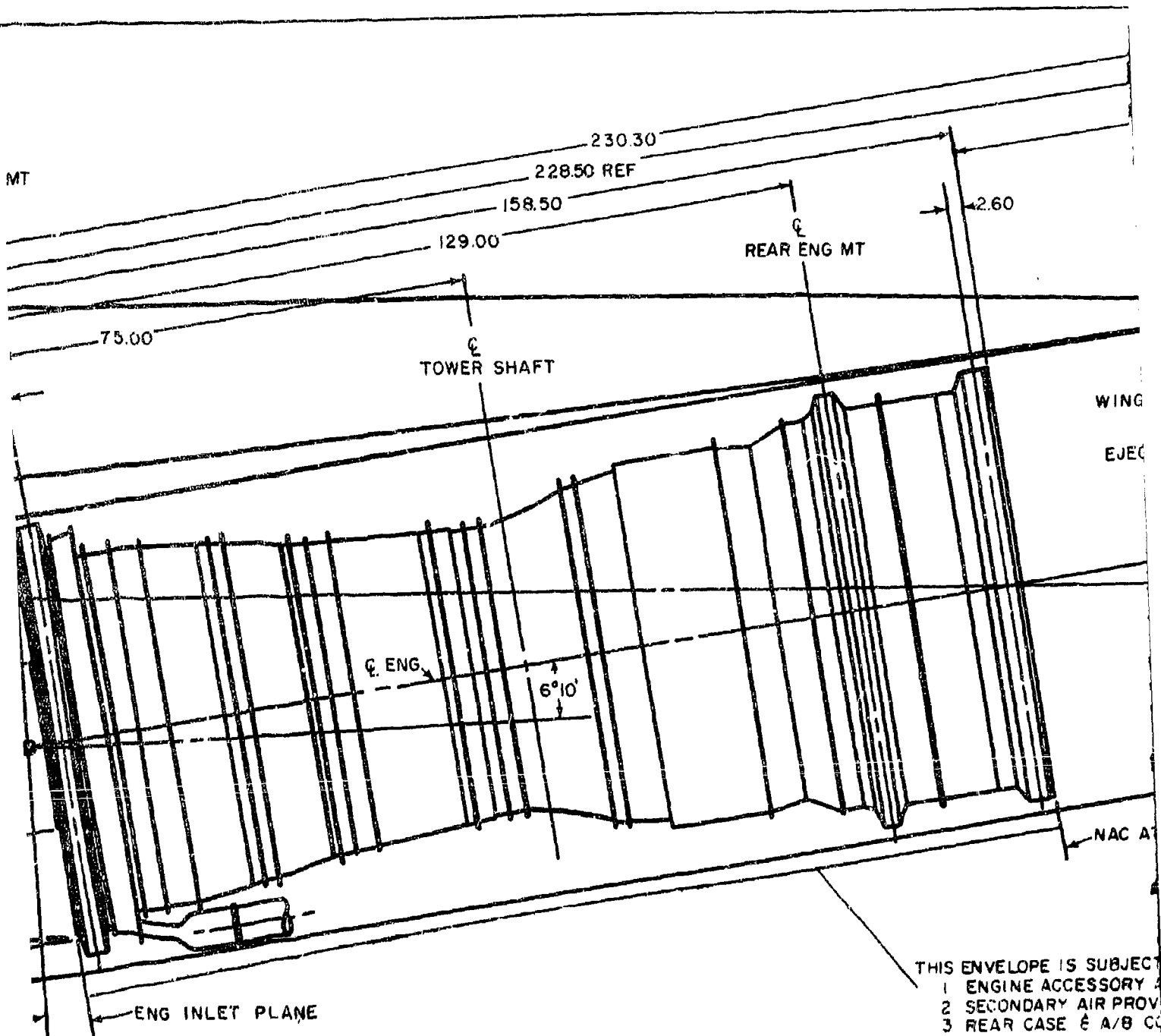
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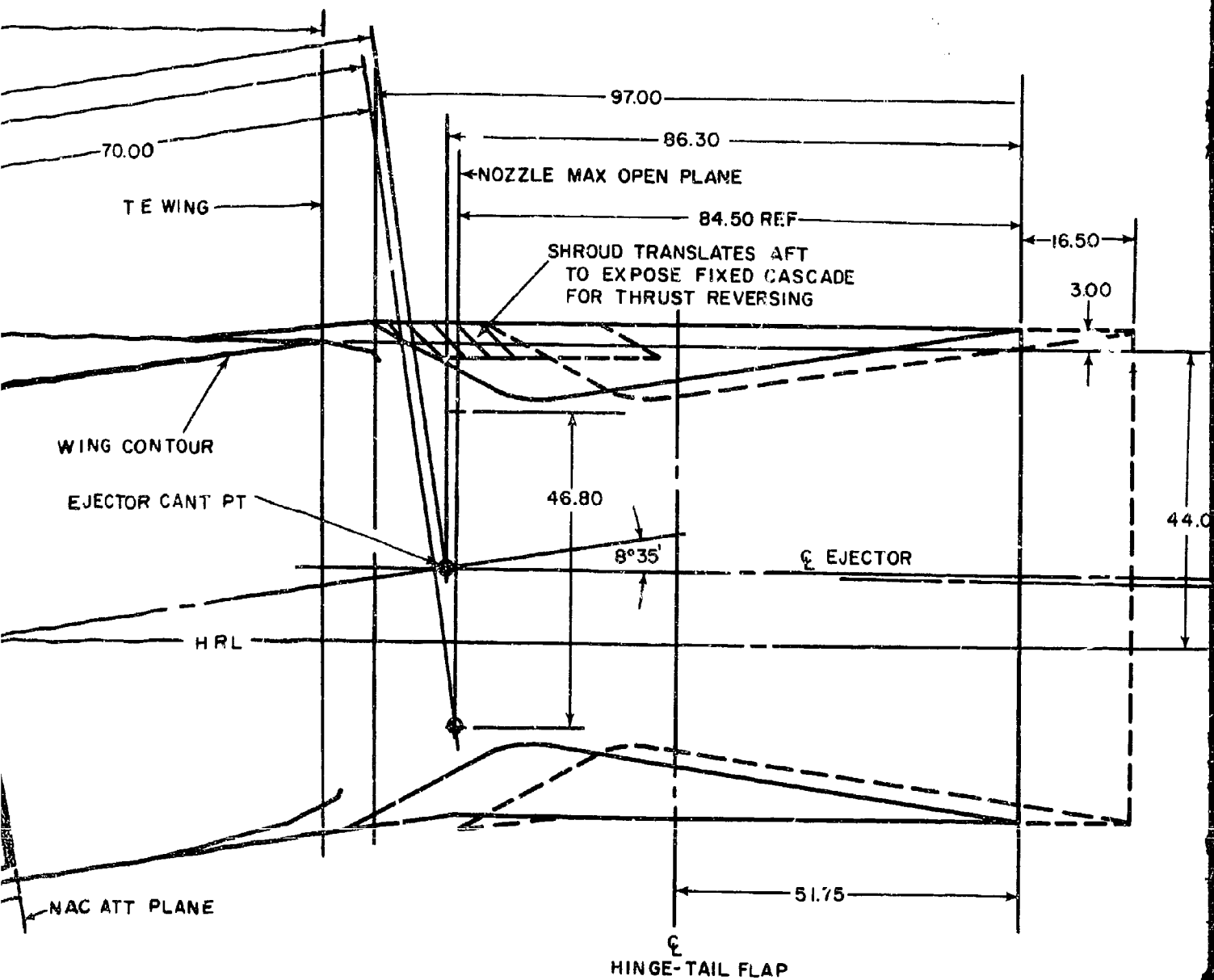
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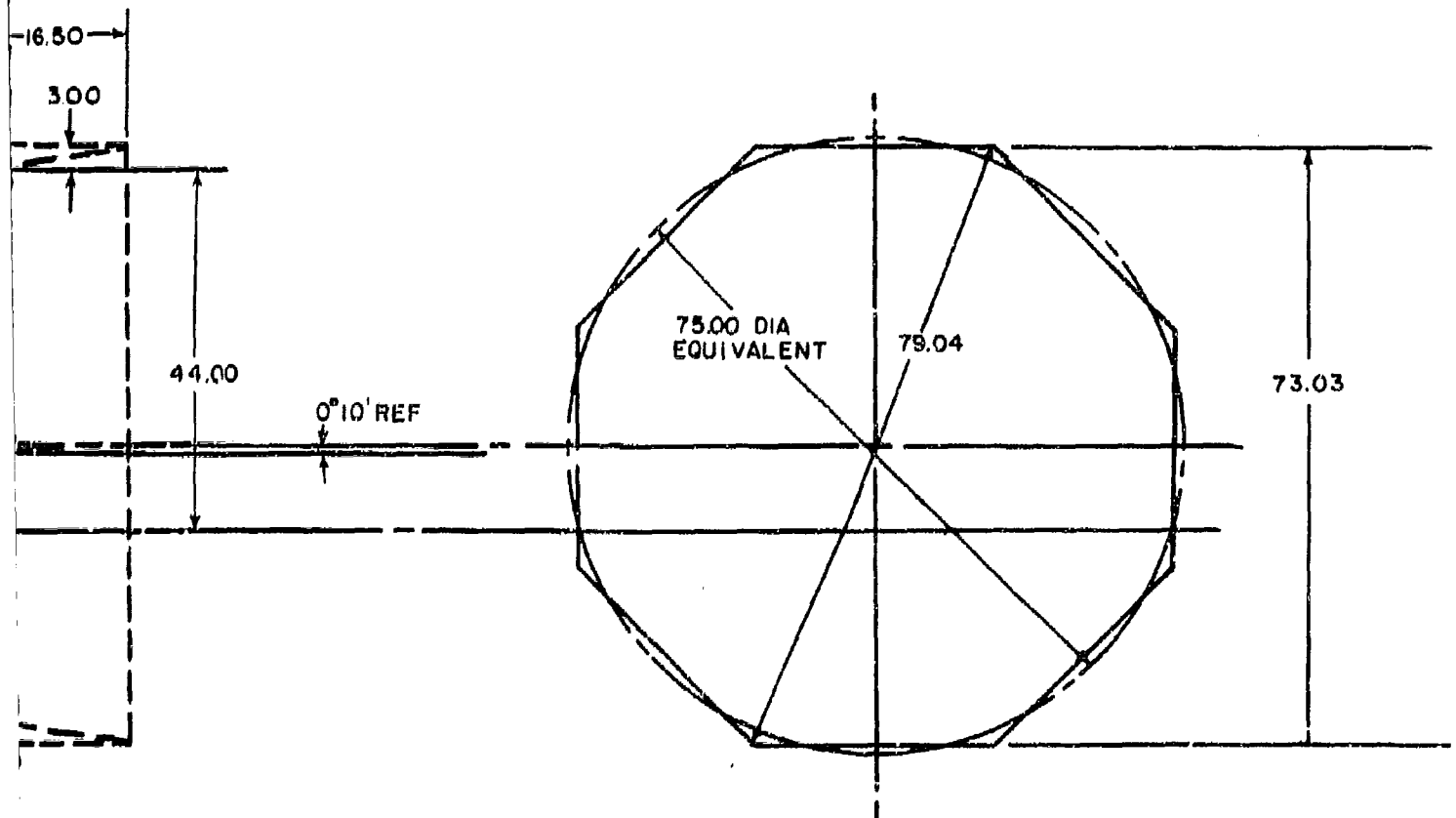




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 ACCESSORY ARRANGEMENT  
 AIR PROVISION FOR EJECTOR  
 SE & A/B COOLING PROVISION

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2000°F TURBINE INLET TEMPERATURE  
L. H. SIDE VIEW OUTBOARD POD  
THIS ENVELOPE IS SUBJECT TO CHANGE FOR:  
1. ENGINE ACCESSORY ARRANGEMENT  
2. SECONDARY AIR PROVISION FOR EJECTOR  
3. REAR CASE AND A/B COOLING PROVISION

STJ227, 525 LBS./SEC. (BASE FLOW) TURBOJET

Figure 1-22

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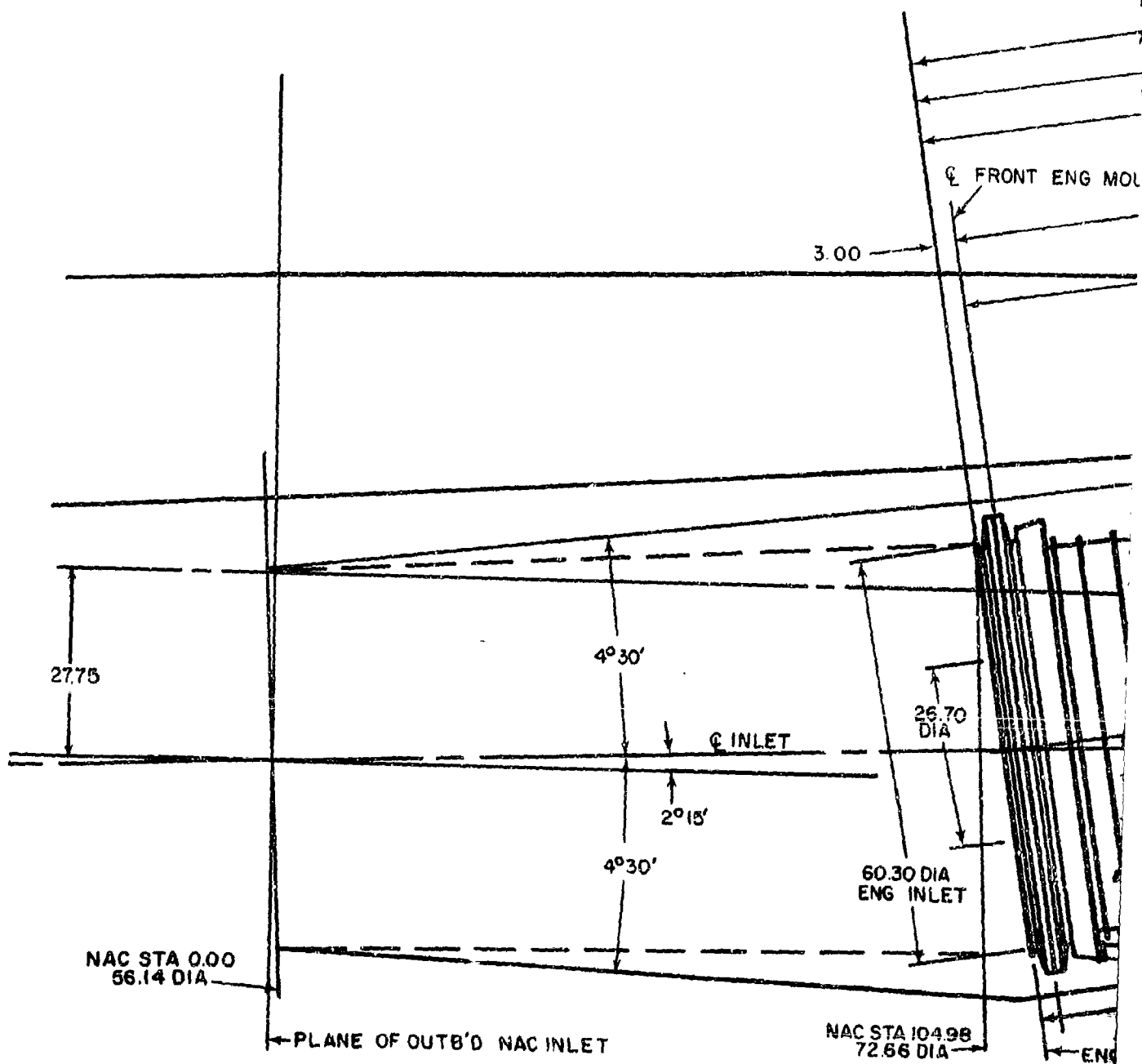
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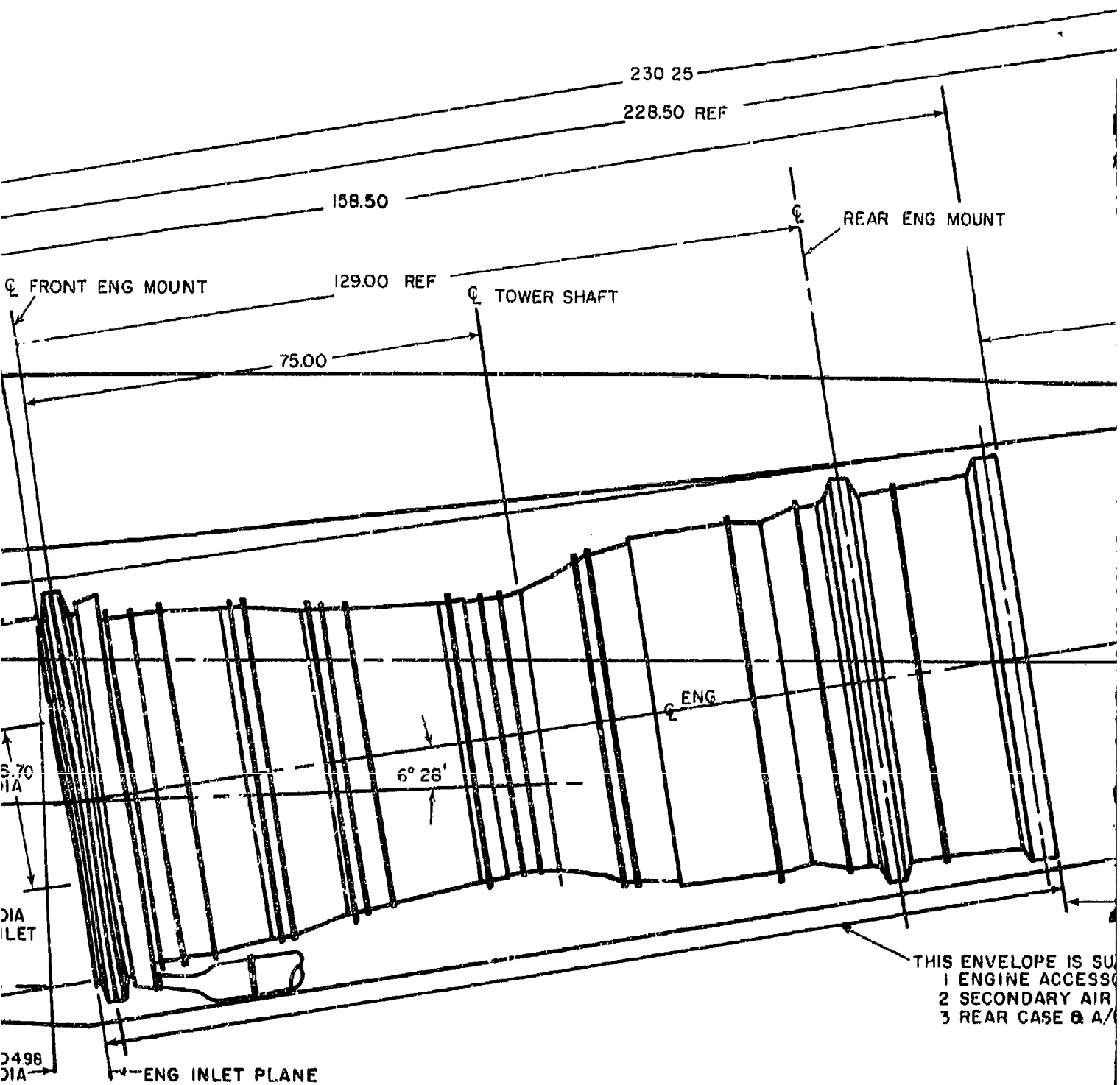
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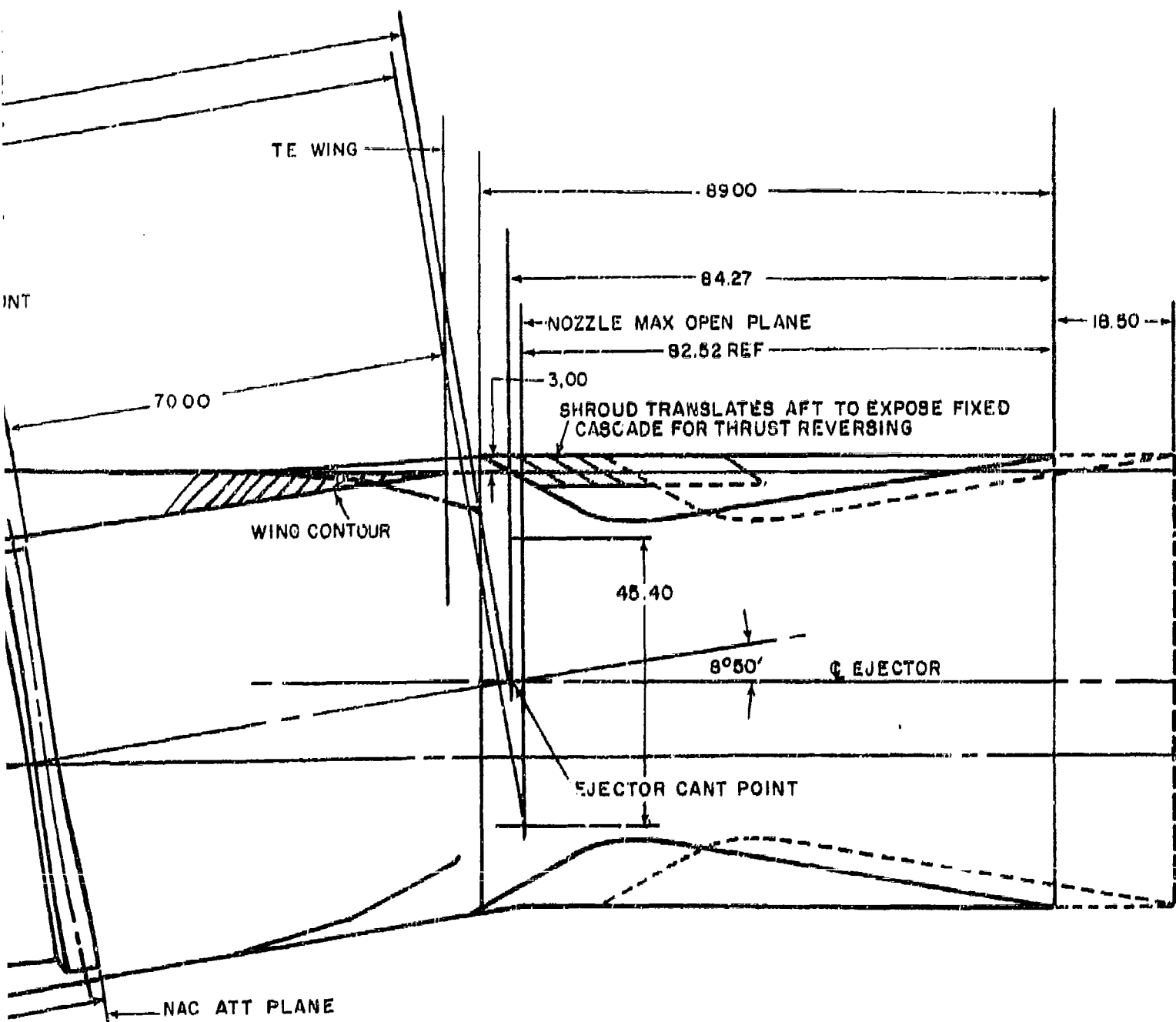
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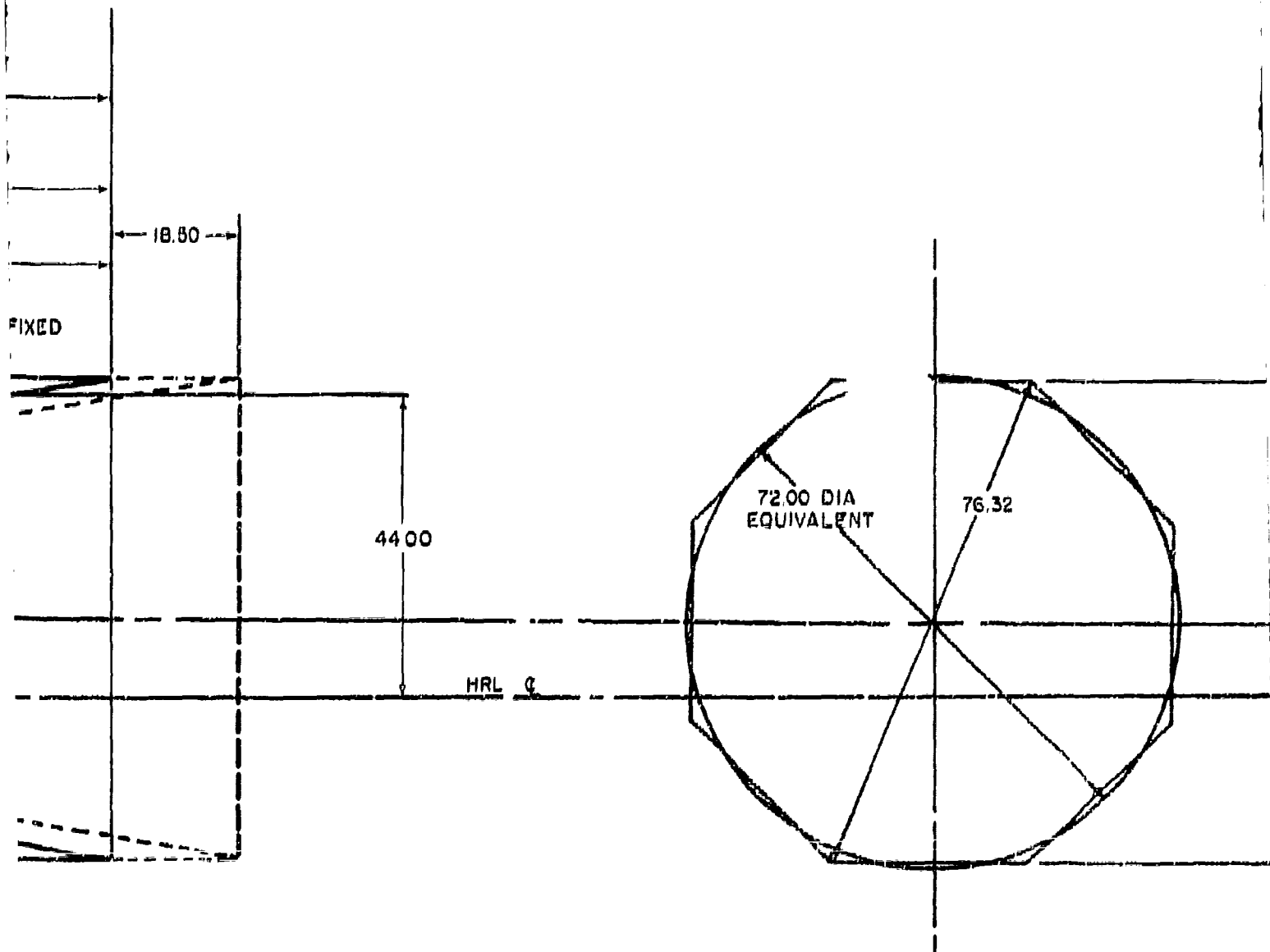






ENVELOPE IS SUBJECT TO CHANGE FOR  
ENGINE ACCESSORY ARRANGEMENT  
SECONDARY AIR PROVISIONS FOR EJECTOR  
NEAR CASE & A/B COOLING PROVISIONS

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2000°F TURBINE INLET TEMPERATURE  
THIS ENVELOPE IS SUBJECT TO CHANGE FOR:  
1. ENGINE ACCESSORY ARRANGEMENT  
2. SECONDARY A/R PROVISIONS FOR EJECTOR  
3. REAR CASE & A/B COOLING PROVISIONS

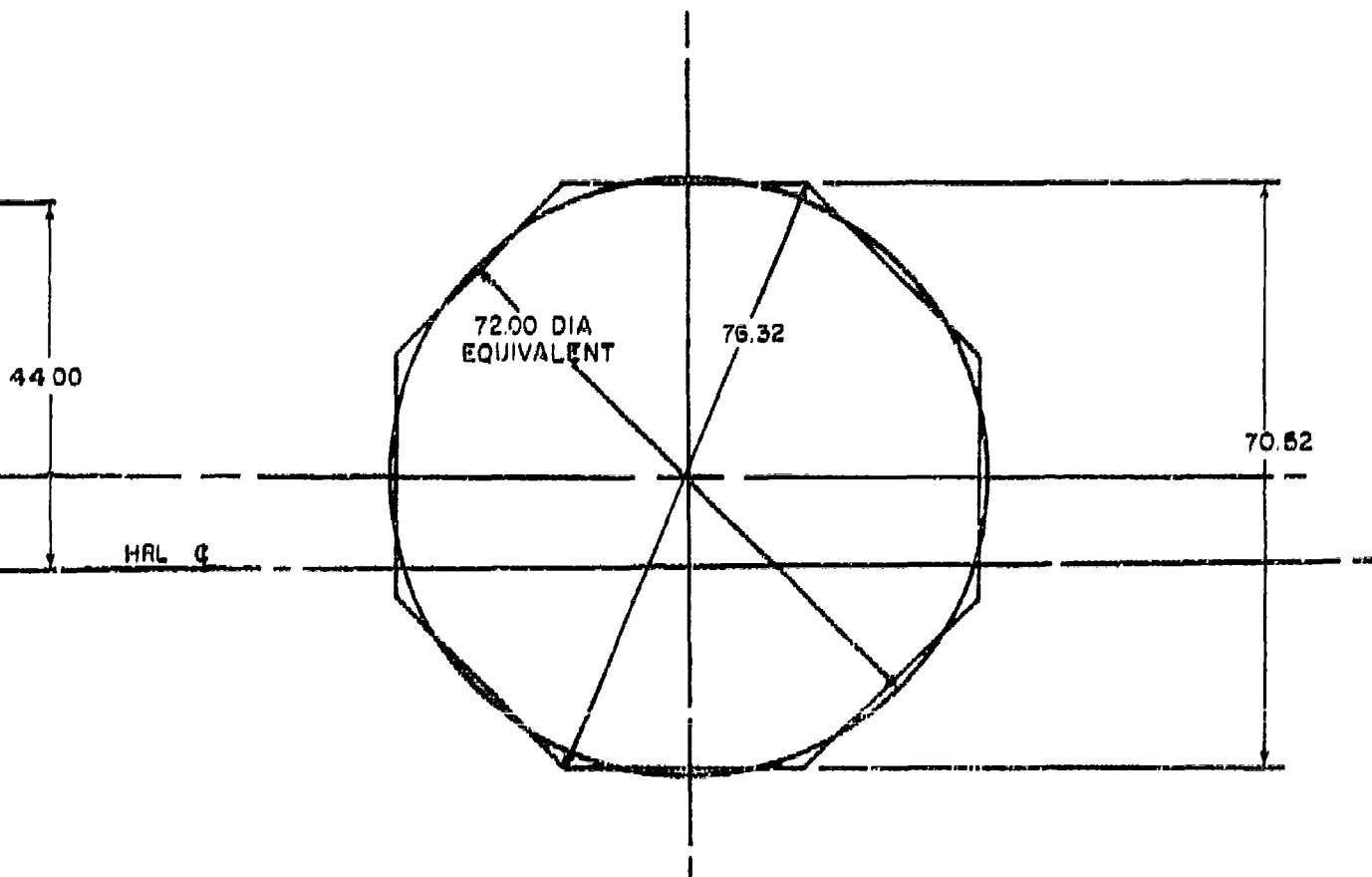
STJ227, 525 LBS./SEC. (LOW FLOW) TURBOJET

Figure 1-23

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1. ENGINE ACCESSORY ARRANGEMENT  
2. SECONDARY A/R PROVISIONS FOR EJECTOR  
3. REAR CASE & A/B COOLING PROVISIONS

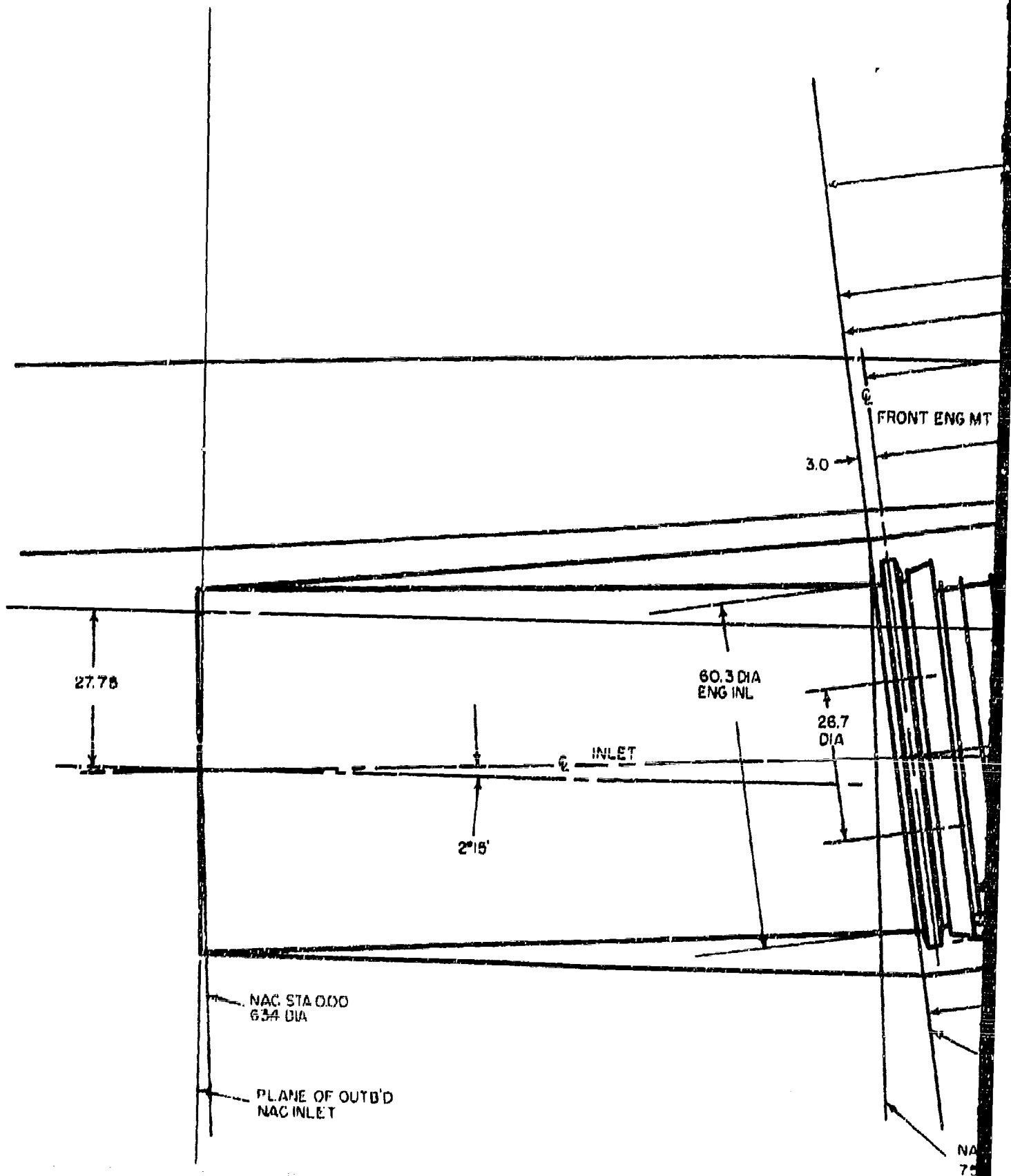
STJ227, 525 LBS./SEC. (LOW FLOW) TURBOJET

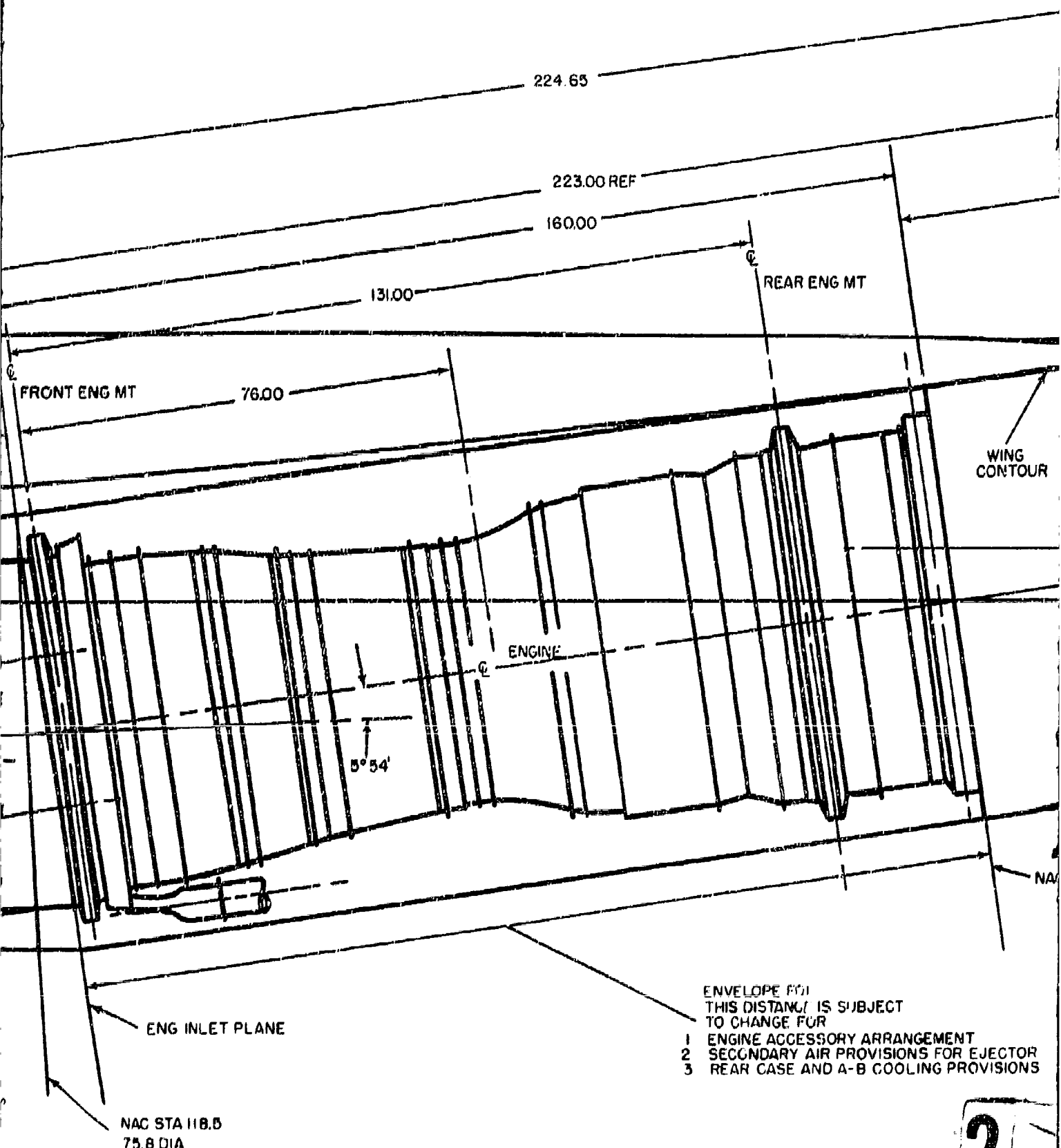
Figure 1-23

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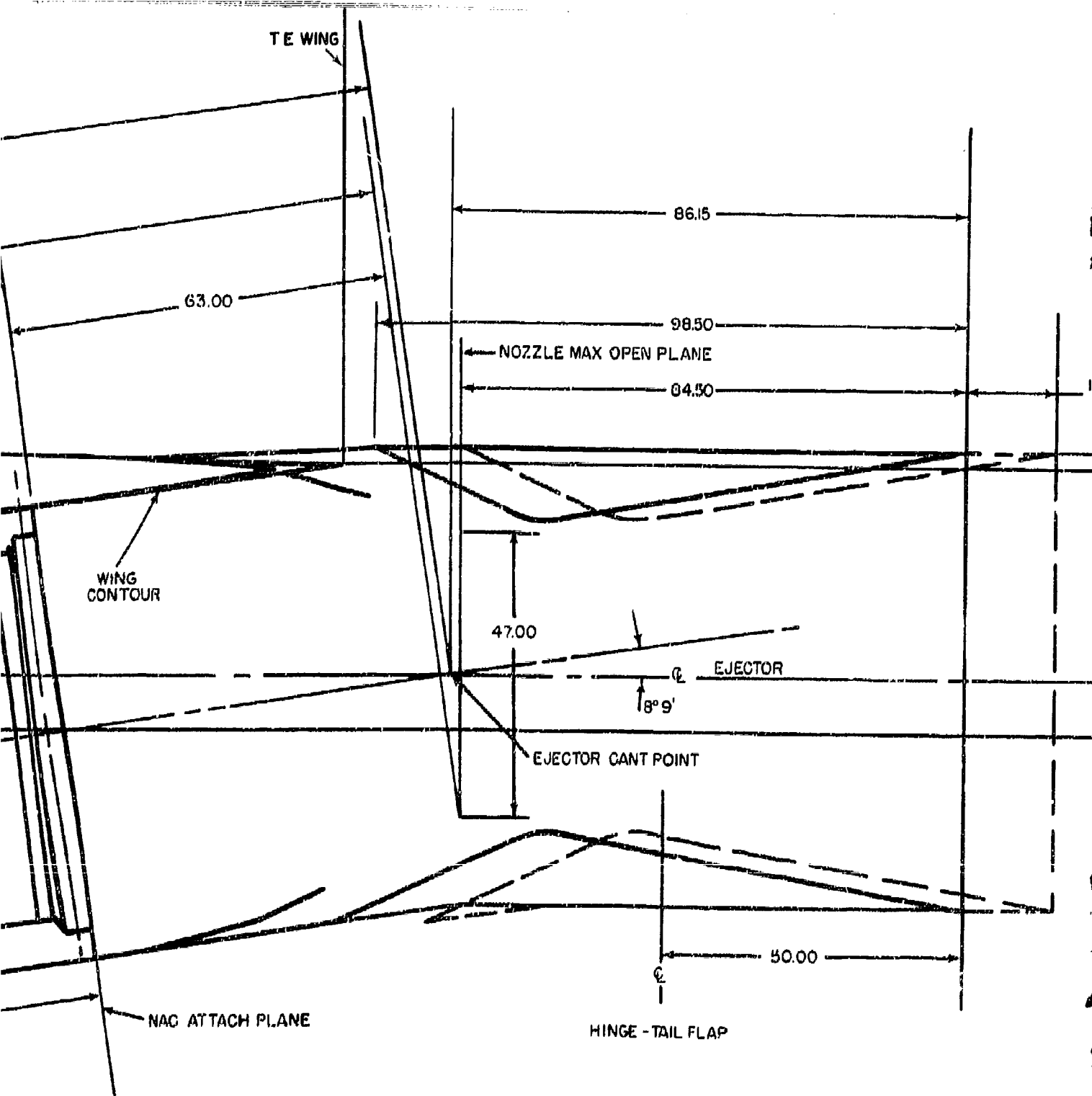
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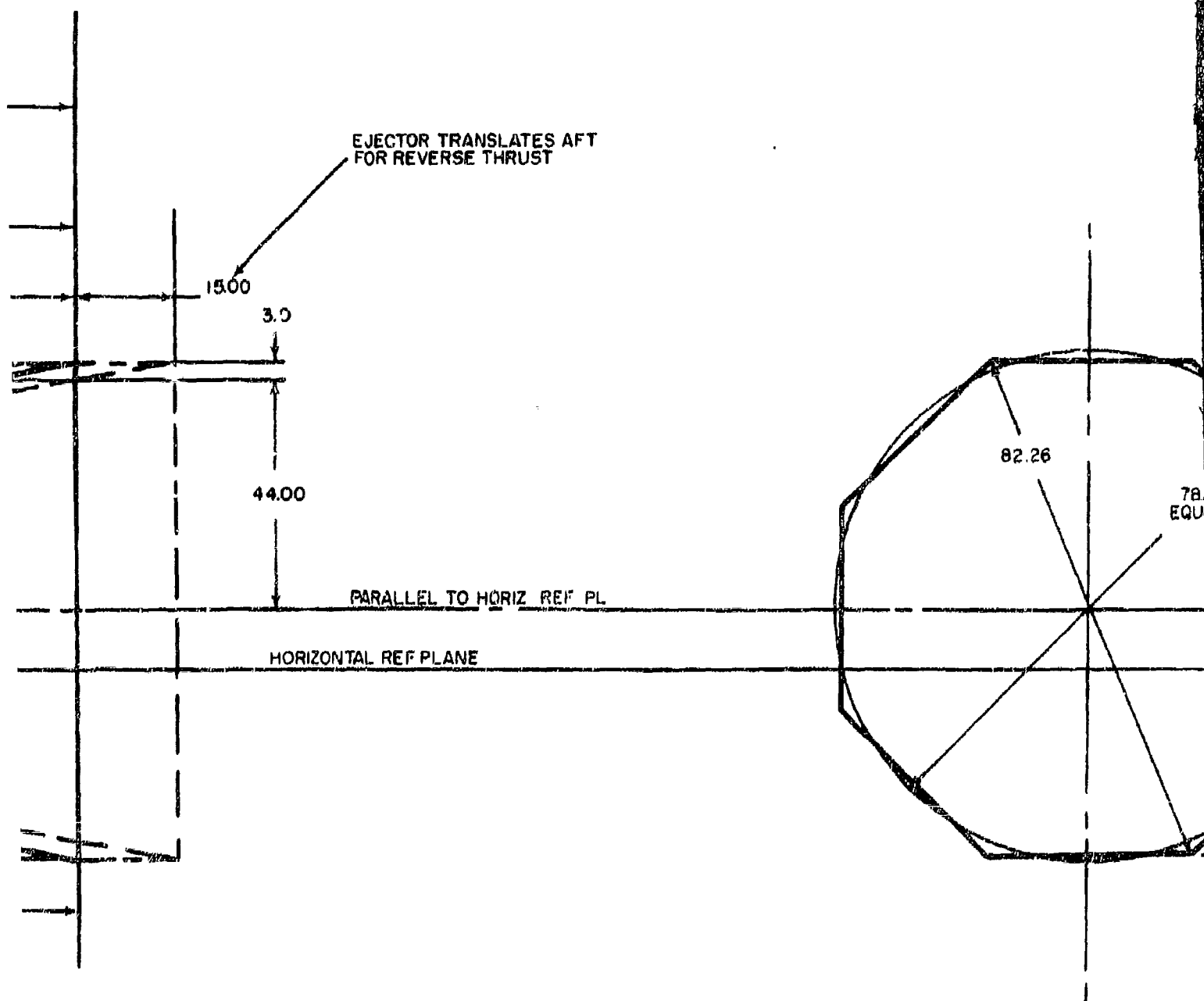
- 1 ENGINE ACCESSORY ARRANGEMENT
- 2 SECONDARY AIR PROVISIONS FOR EJECTOR
- 3 REAR CASE AND A-B COOLING PROVISIONS



ST  
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 OLING PROVISIONS



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2300°F TURBINE INLET TEMPERATURE  
STJ227, 525 LBS./SEC. (HIGH FLOW)

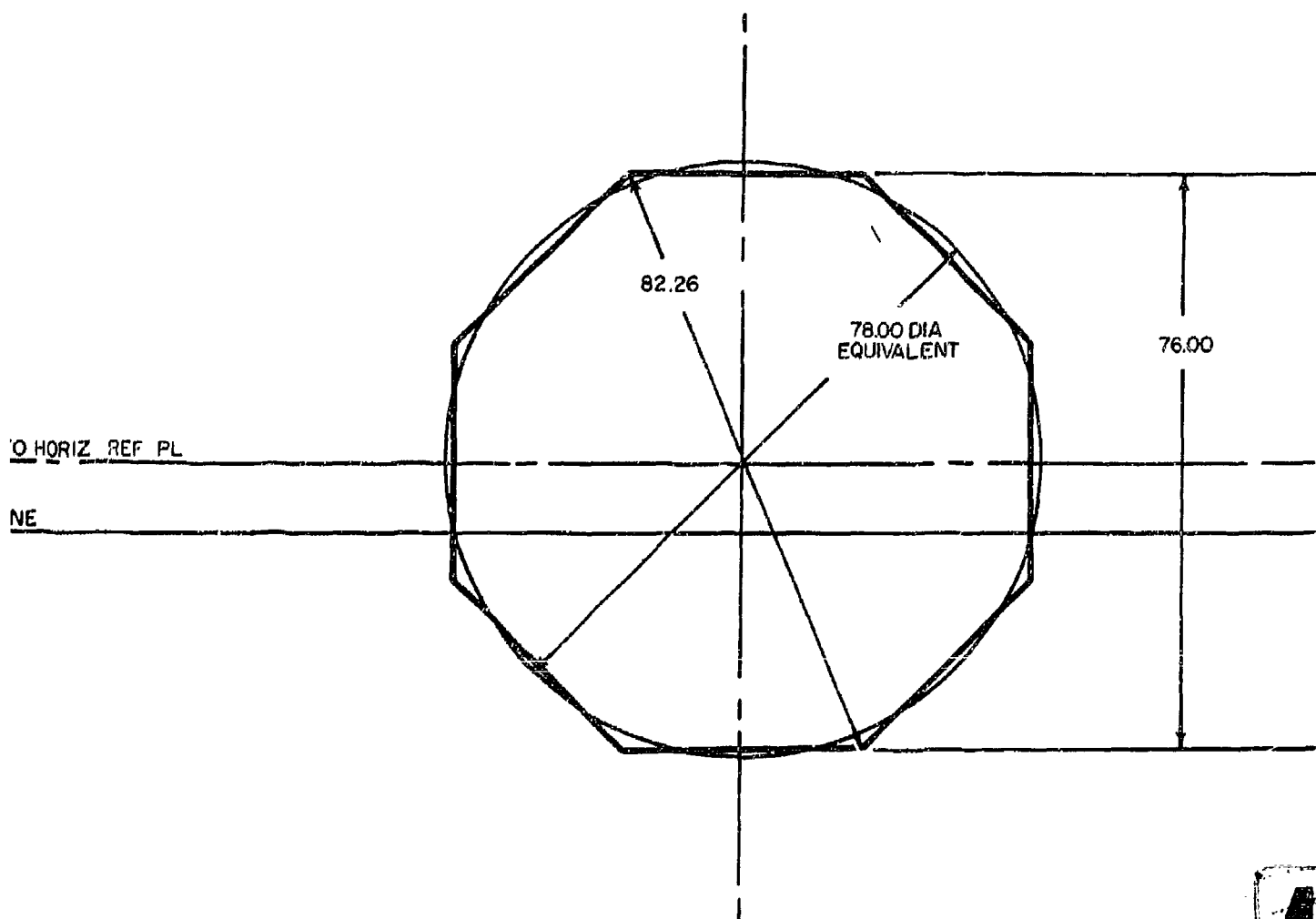
Figure 1-24

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2300°F TURBINE INLET TEMPERATURE  
STJ227, 525 LBS./SEC. (HIGH FLOW) TURBOJET

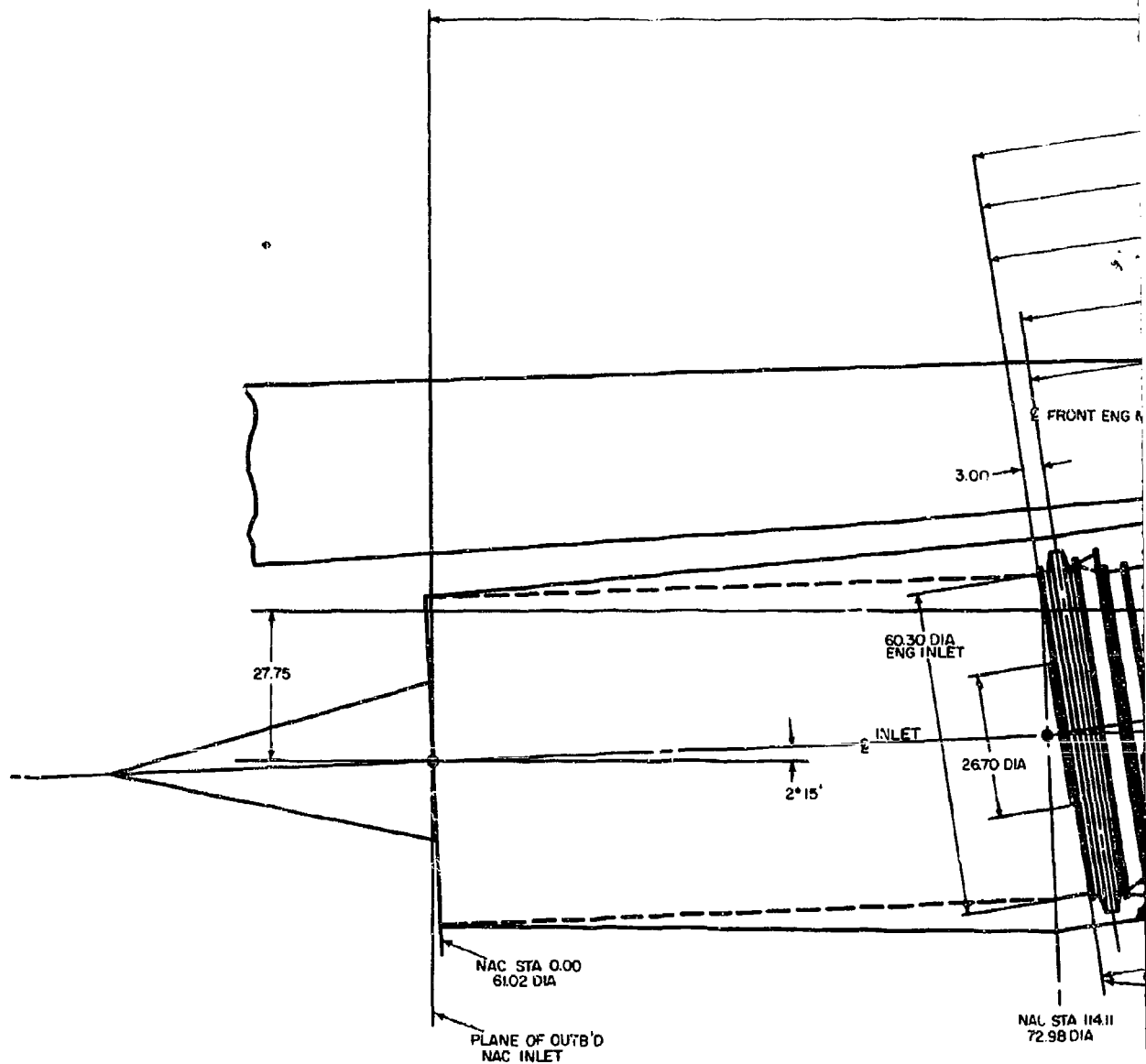
Figure 1-24

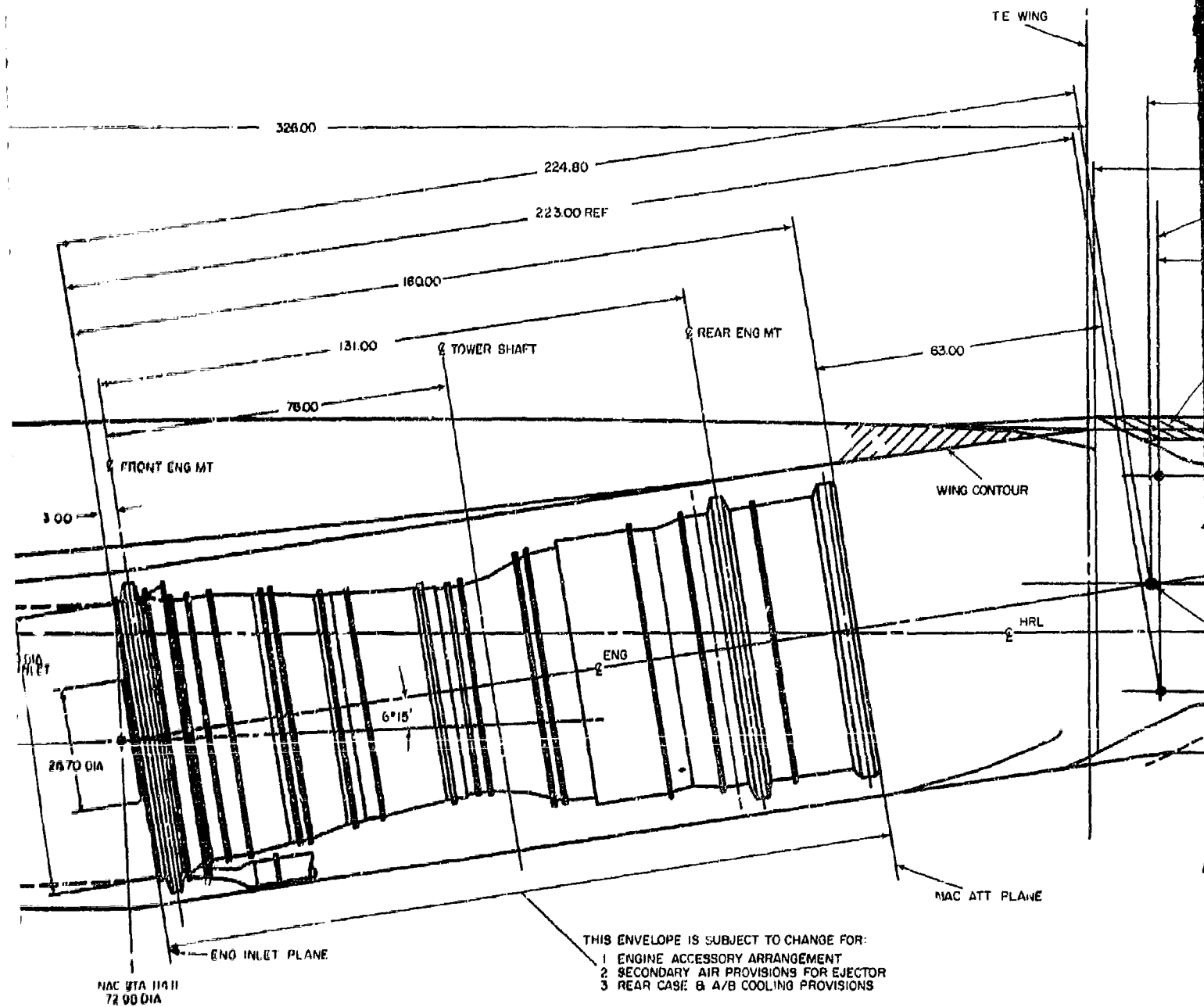
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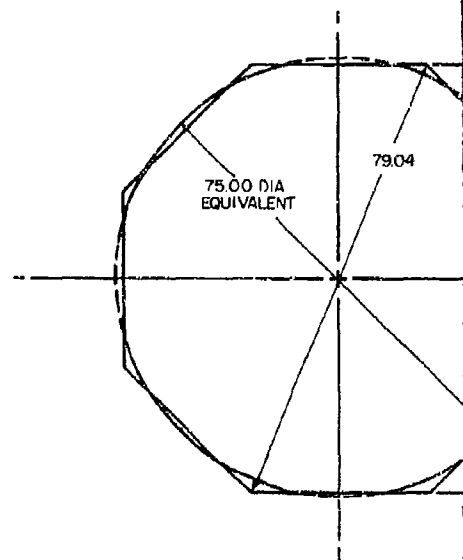
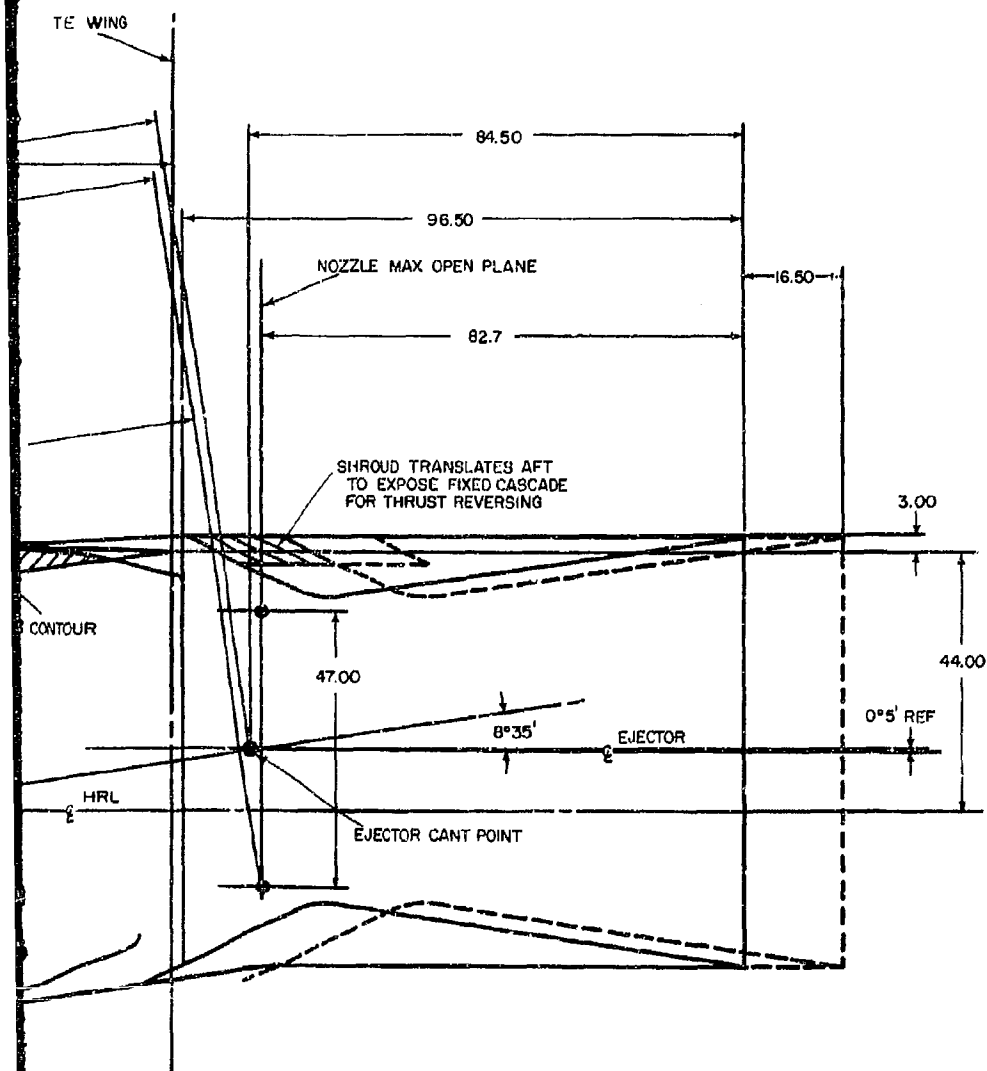
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1. ENGINE ACCESSORY ARRANGEMENTS  
2. SECONDARY AIR PROVISIONS FOR  
3. REAR CASE & A/B COOLING PROVISIONS

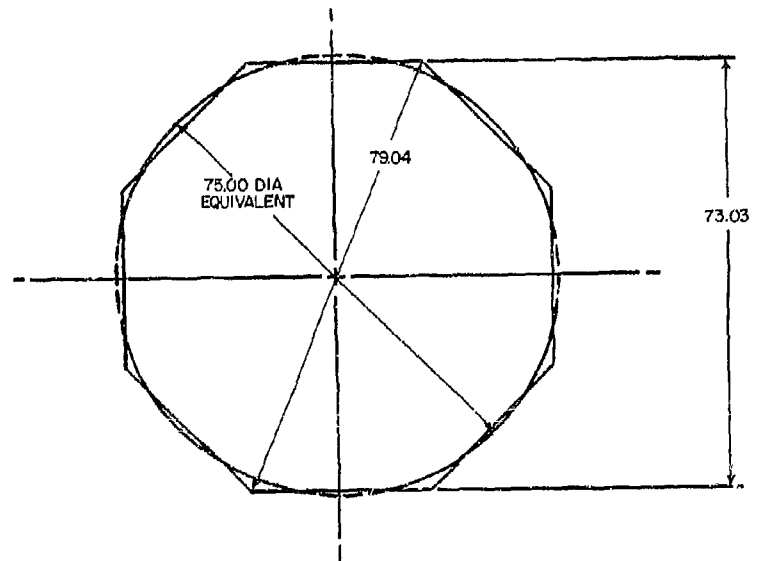
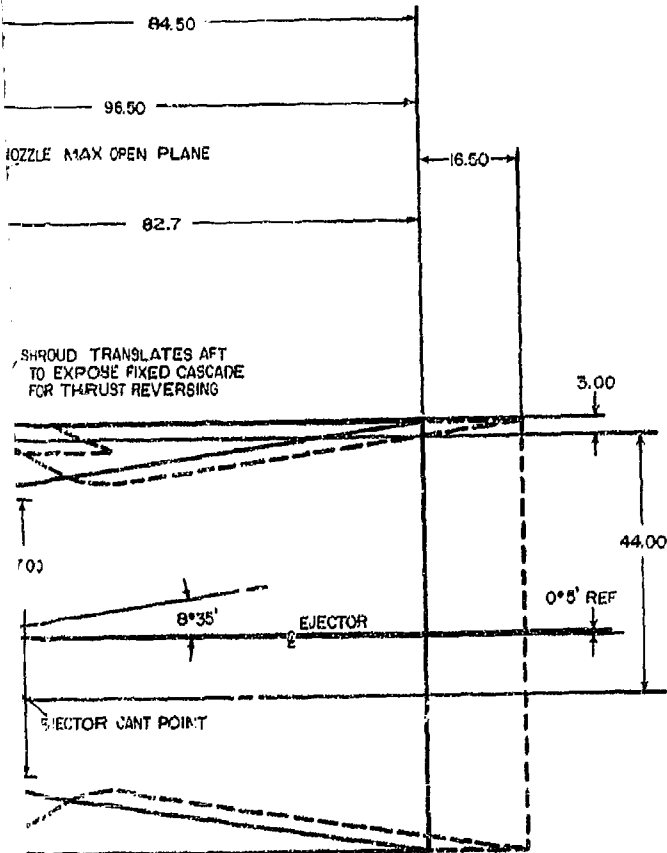
STJ227, 525 LBS./SEC. (BASE FLOW) TURBOJET

Figure 1-25

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2300°F TURBINE INLET TEMPERATURE  
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1. ENGINE ACCESSORY ARRANGEMENT  
2. SECONDARY AIR PROVISIONS FOR EJECTOR  
3. REAR CASE & A/B COOLING PROVISIONS

STJ227, 525 LBS./SEC. (BASE FLOW) TURBOJET

Figure 1-25

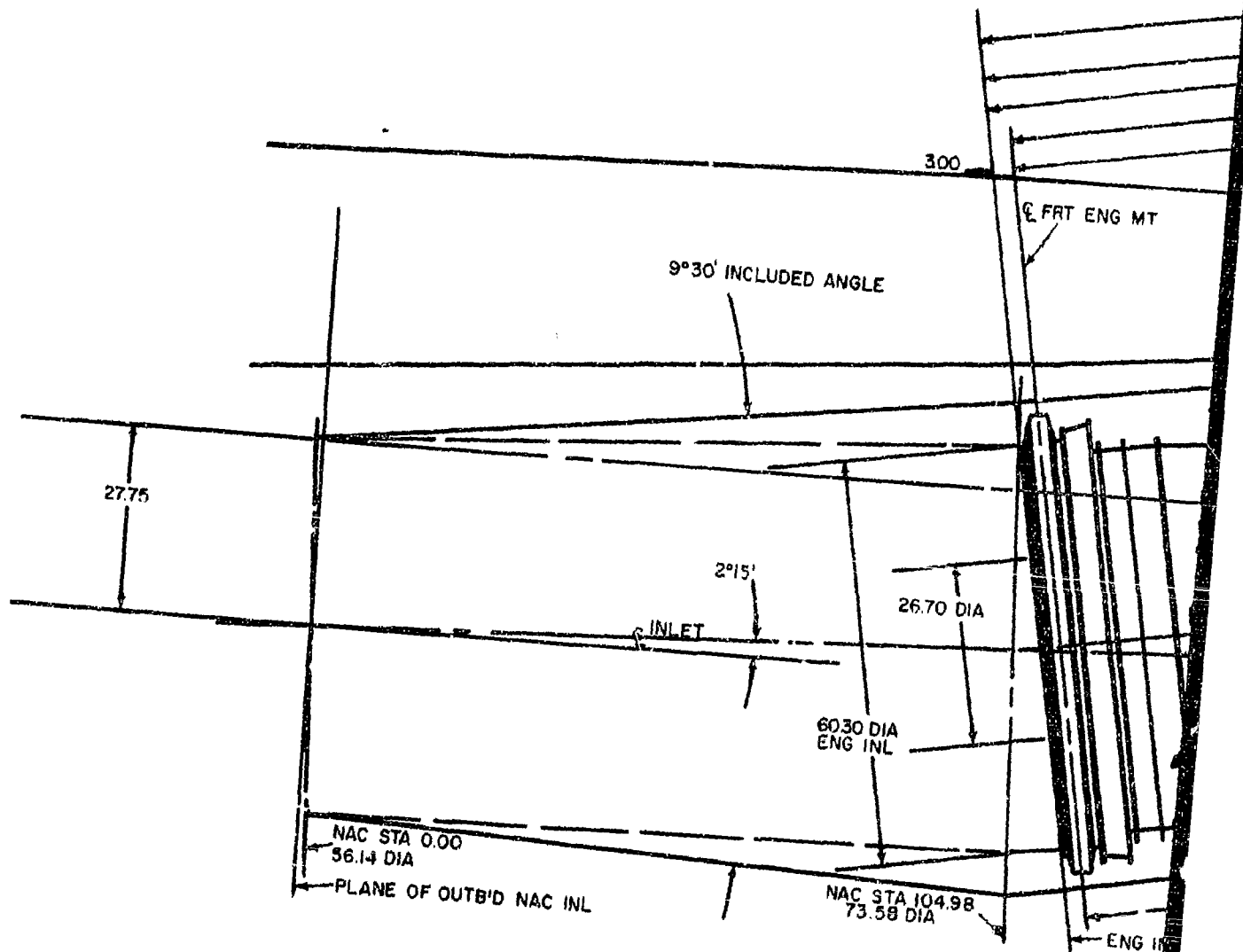
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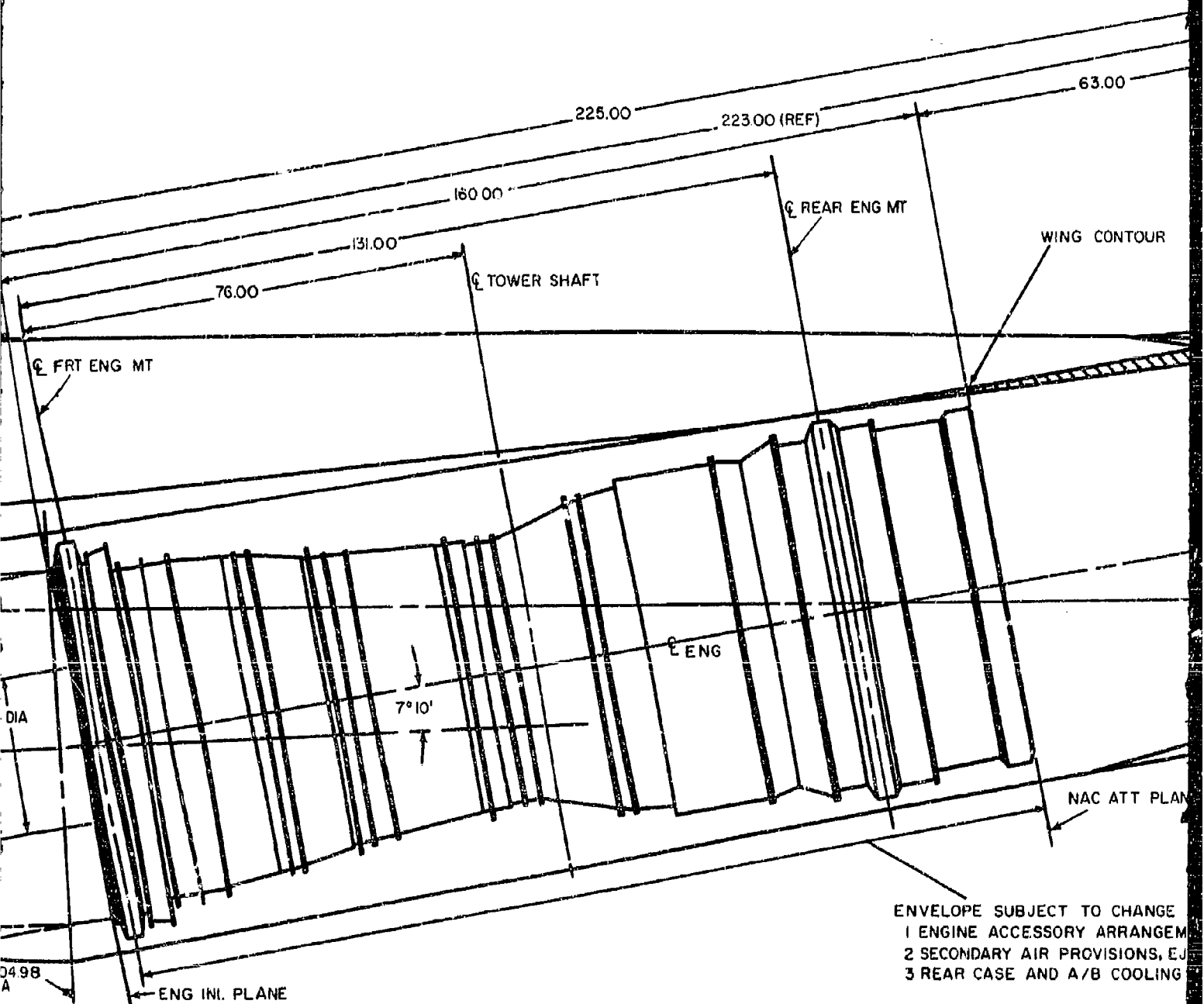
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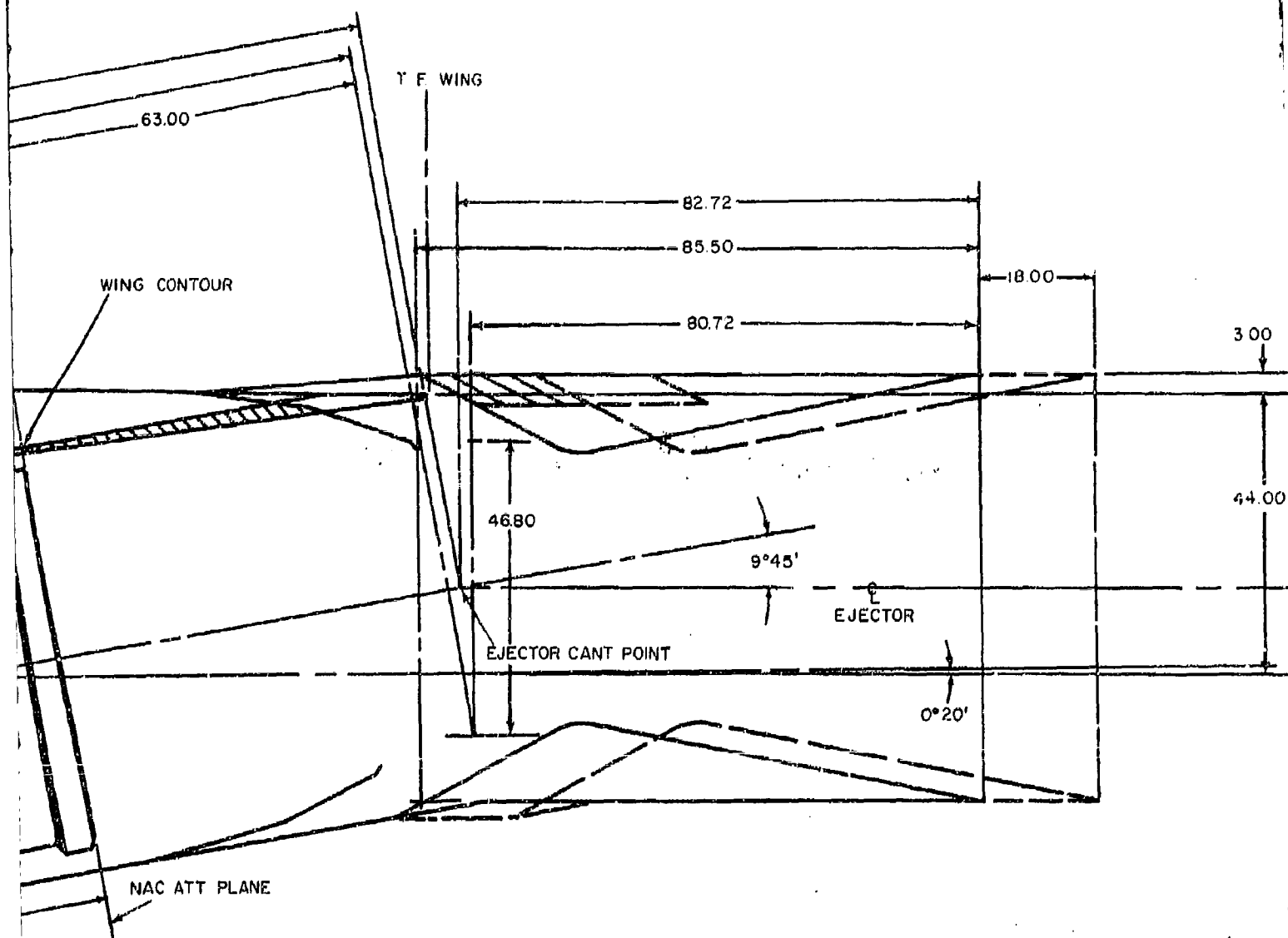
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PE SUBJECT TO CHANGE  
 IE ACCESSORY ARRANGEMENT  
 IDARY AIR PROVISIONS, EJECTOR  
 CASE AND A/B COOLING PROVISIONS

PWA-2600



Figure 1-26

1. 在下列各题中，选择正确的答案，将序号填入括号内。

ACCESSORY DRIVE FROM

POWER TAP OFF  
BLOWMETER LITE PLO 241 (144) 20002178  
HYPERMILL K. PLUMB (1704) 145 4482

**CONCLUSION:** Contraceptive use was

COMBUSTION: CHAMBER PLE DRAIN  
FLAP VALVE DRAIN (MAIN)  
FLAP VALVE DRAIN (A/B ENGINE)  
FLAP VALVE DRAIN (A/B ENGINE)  
HYDRAULIC COMBUSTION CHAMBER PLE DRAIN

### CONCLUSIONS

FUEL PUMP INLET PRESSURE  
ORION VALVE FUEL PRESSURE OUTLET PRESSURE

AD. DATA ACQUISITION CARD

WASH: FUEL FLOWMETER SUPPLY PLEI  
WASH: FUEL FLOWMETER SUPPLY OUTL/I  
AFTTHROUWER FLOWMETER SUPPLY PLEI  
AFTTHROUWER FLOWMETER SUPP. OUTL/I

Figure 1. The effect of the concentration of the polymer on the gelation time.

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NUMBER OF THE PAGES

4. **WPF CUM**

1. TYP. OVERFLOW GRAB  
 2. CUP OVER THE GRAB  
 3. RANDOM WALK OR DRAG  
 4. STRANDY GRAB

• **Stress** – the body's response to any demand or challenge

YOUNG PEOPLE'S FULL-TIME  
NIGHT SCHOOL

1000

3. 5. 1778 OUTLET PRESSURE ☐ PROVISIONS FOR PRESSURE  
DIFFERENTIAL

1. **THEORY**

07 06 1964

**Abstract** The purpose of this study was to determine the effect of a 12-week training program on the physical fitness of 10-year-old children. The study was conducted in a primary school in Ankara, Turkey. The study group consisted of 20 children (10 boys and 10 girls) who were randomly selected from the 10-year-old children in the school. The children were divided into two groups: a control group and an experimental group. The control group did not participate in any physical activity program, while the experimental group participated in a 12-week training program. The physical fitness of the children was measured at the beginning and at the end of the 12-week period. The measurements included maximum heart rate, maximum oxygen consumption, maximum power, and maximum speed. The results of the study showed that the experimental group had significantly higher values for all four measurements at the end of the 12-week period compared to the control group. The results suggest that a 12-week training program can improve the physical fitness of 10-year-old children.

4-11-68 104-2071A 2027

2000 年 12 月 15 日

1. CONTROL SEAL CRASH  
2. PLUMP SEAL CRASH  
3. PLUMP SEAL CRASH  
4. PLUMP SEAL CRASH  
5. PLUMP SEAL CRASH

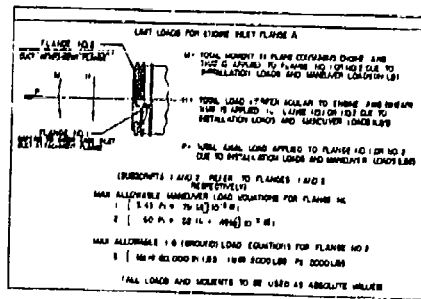
and fuel management, among others.

TEMPERATURE / FOMENAL

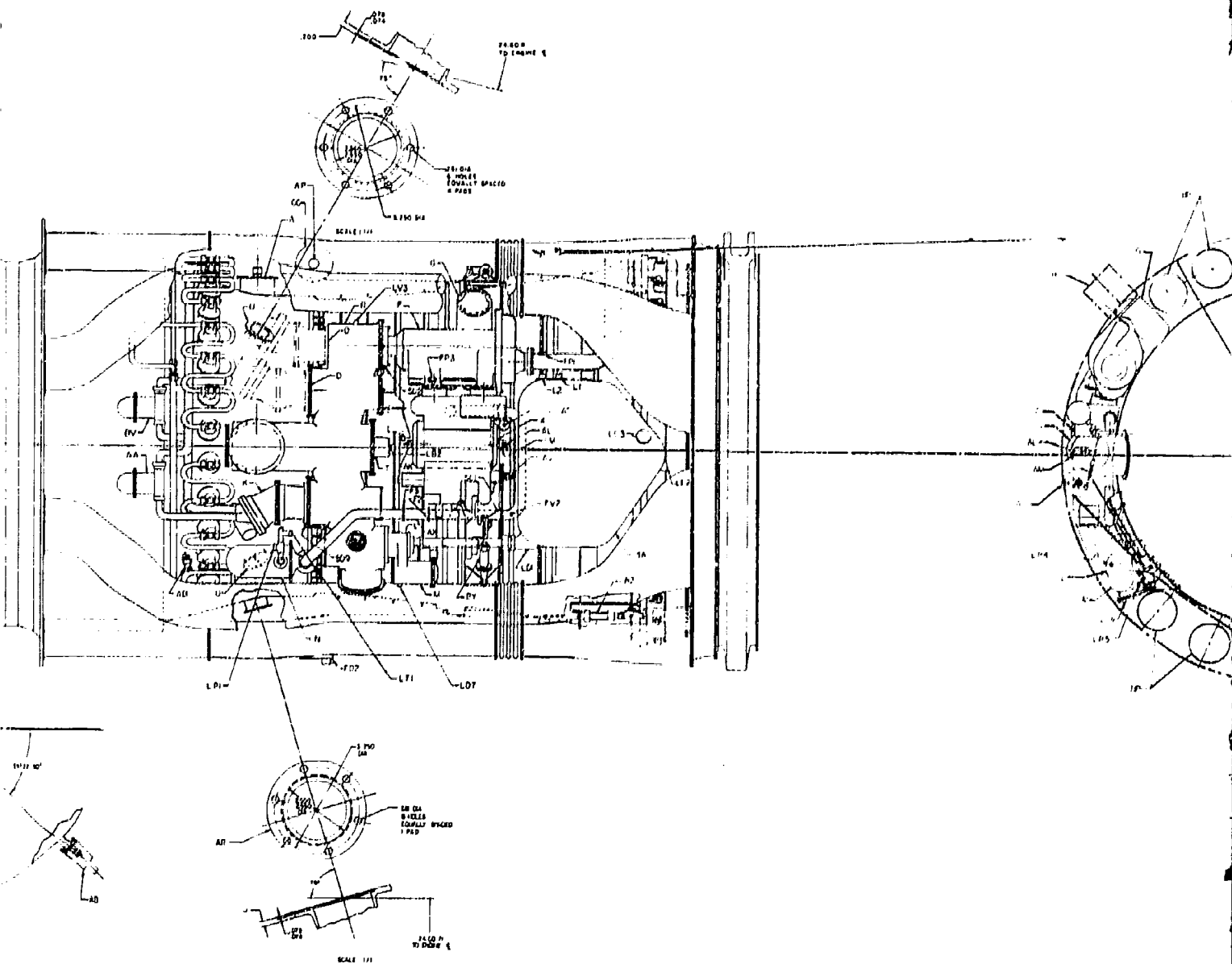
**Abstract**

FUEL PUMP  
PUMP FUEL FILTER (HASS)  
PUMP FILTER (1/4" SPACE FOR REMOVAL)  
REPLICA FUEL PUMP  
AUXILIARY PUMP  
FILTER  
OIL COOLER (HASS)  
OIL COOLER (1/4" TOLERANCE)  
BURN  
BURNER RESTART POSITION  
THERMALLY PROTECTED VALVE  
PRESSURE DUE TO PRO  
FUEL CONTROL  
PRESSURE FUEL CONTROL

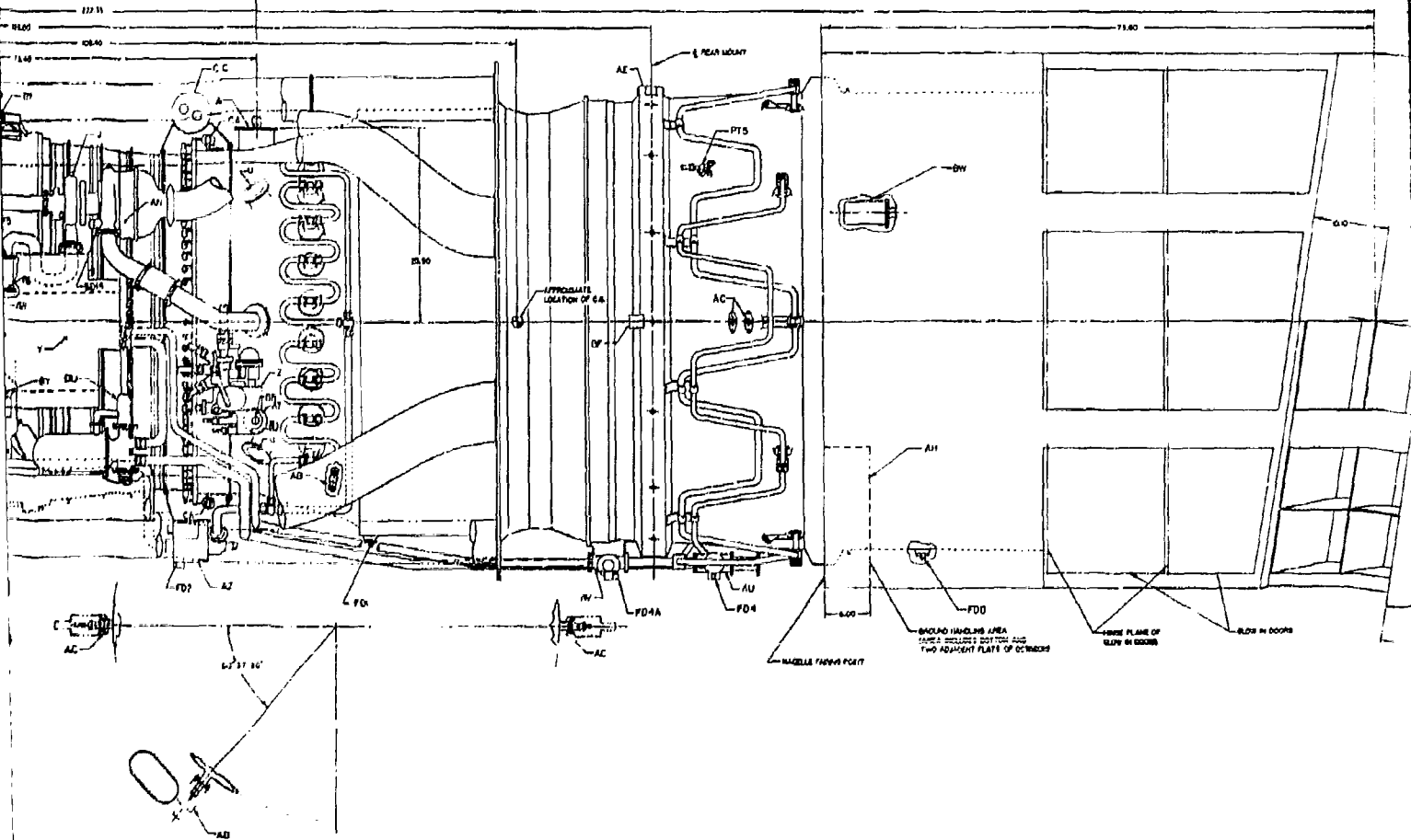
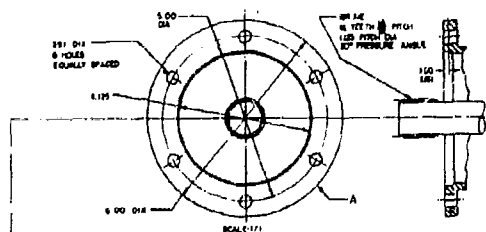
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111	101	X
111	101	Y

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C. 1007, 11-11-11







VIEW OF MAIN ENGINE  
LOOKING FORWARD

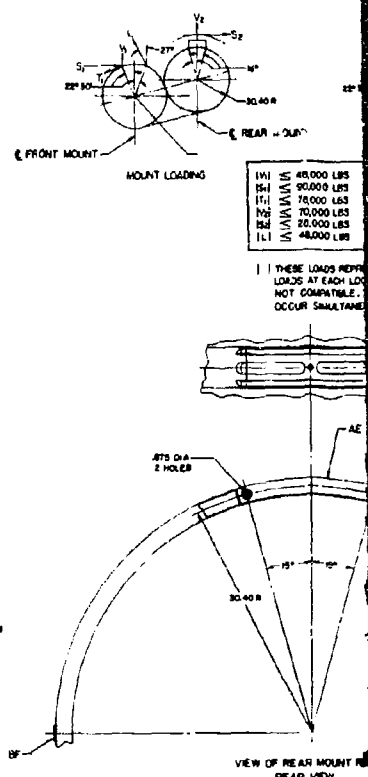
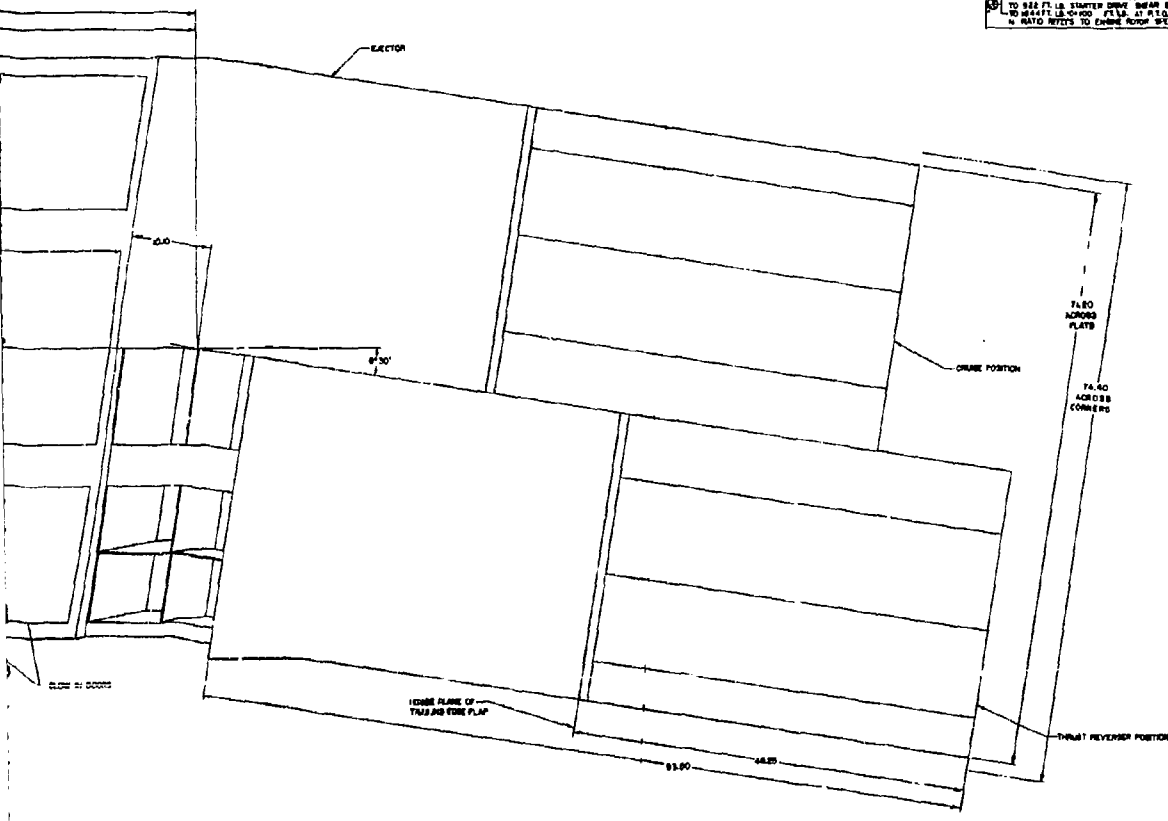
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TORQUE & MOMENT LIMITATION TABLE-ACCESS DRIVES							
DRIVE PNO	NOMINAL SIZE	TORQUE (L.B.-IN.)			SPEED RPM	ROTATION (FACING ENGINE END)	1 G OVERLOAD (L.D. - 1/2 L.D. IN)
		CONTINUOUS	OVERLOAD	STATIC			
A	TAKEOFF	2334	3500	24000	2,940	CCW	
B	CLIMB	7	50	752	50	CCW	
C	HYDRAULIC	948	1470	4750	980	CCW	

MAX ALLOWABLE CONTINUOUS TORQUE VALUES ARE AT 100% ENGINE SPEED UNLESS OTHERWISE INDICATED. NO DESTRUCTIVE FORCES RESULTING FROM ACCESS VIBRATION OR SHOCK ARE PERMITTED.

MAX ALLOWABLE OVERLOAD BEYOND MOMENTS OF ACCESSORIES ABOUT DRIVE PNO ARE AS SHOWN PROVIDED NO DESTRUCTIVE FORCES RESULTING FROM ACCESS VIBRATION ARE PERMITTED.

MAX ALLOWABLE FOR 5 MINUTE DURATION REQUIREMENTS AT 100% INTERVALS FOR R.T.O. ENGINE SHALL HAVE ADEQUATE STRENGTH TO ACCOMMODATE A MAX TORQUE EQUAL TO 842 FT. LB. SIMILAR DRIVE SECTION SHALL FAIL AT A STATIC TORQUE EQUAL TO 842 FT. LB. WHEN FIELD AT R.T.O. AND IN RATE RETYS TO ENGINE ROTOR WELD.



PROPOSED TURBOJET ACCESSORY

Figure 1-27

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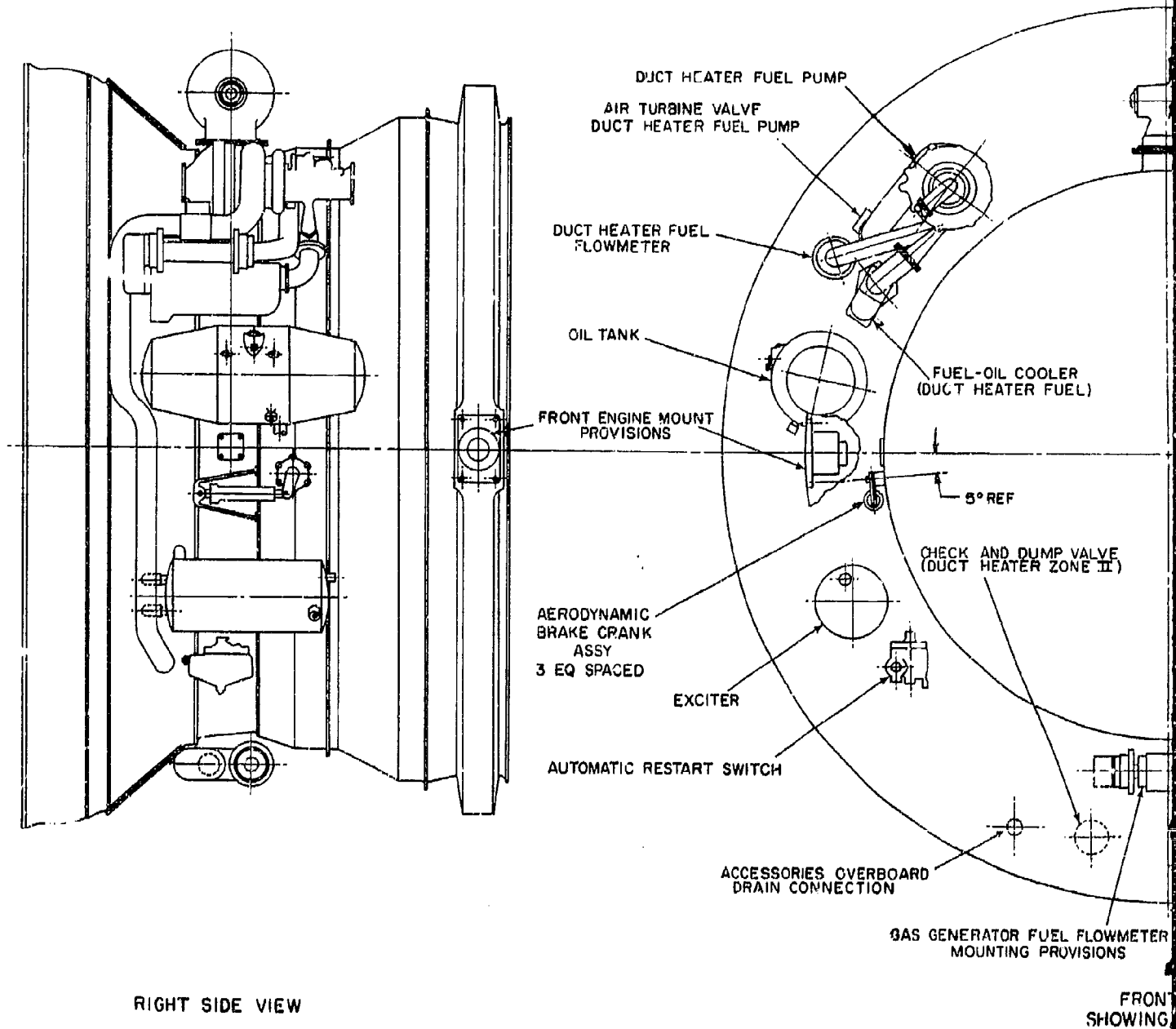
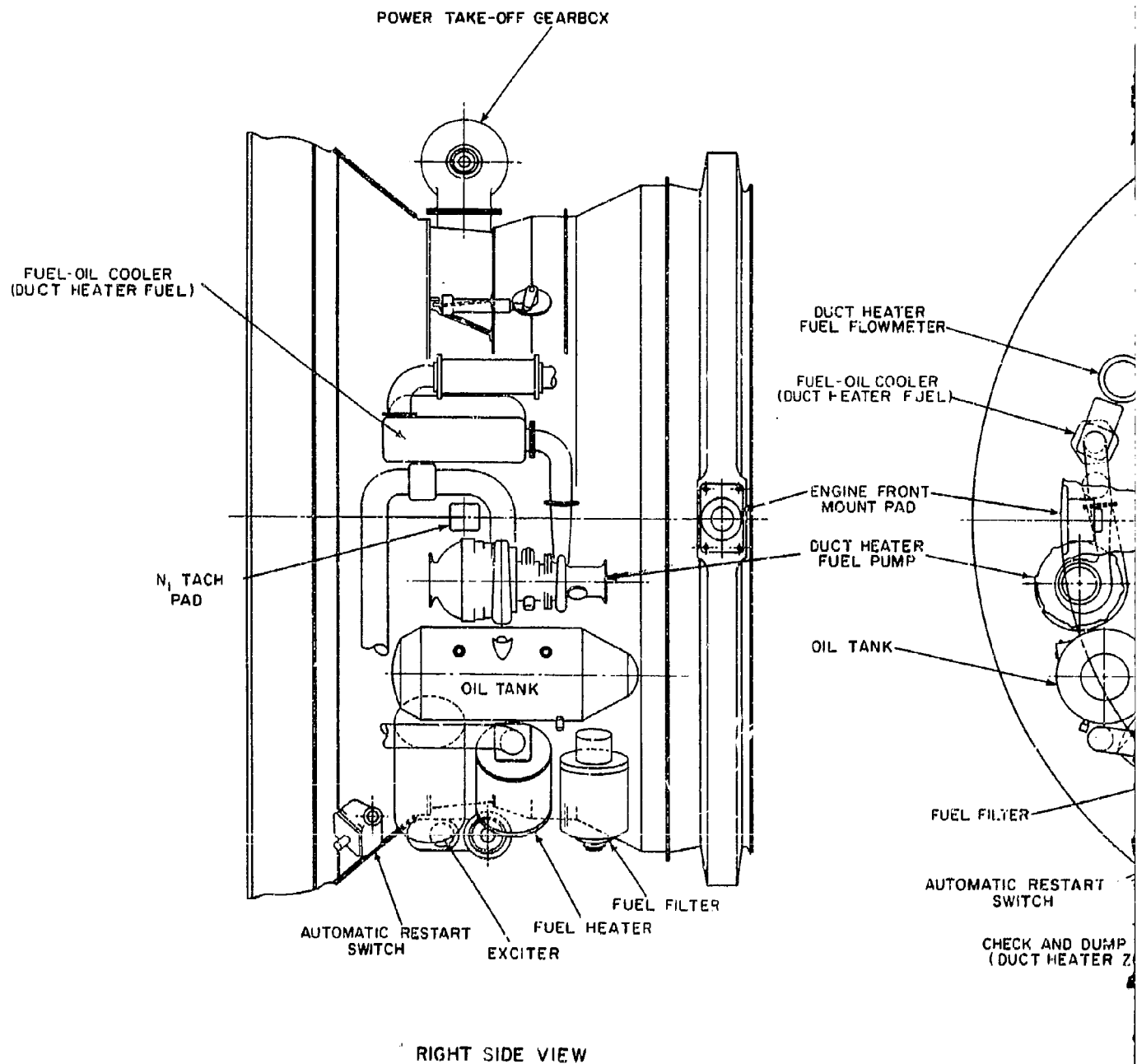
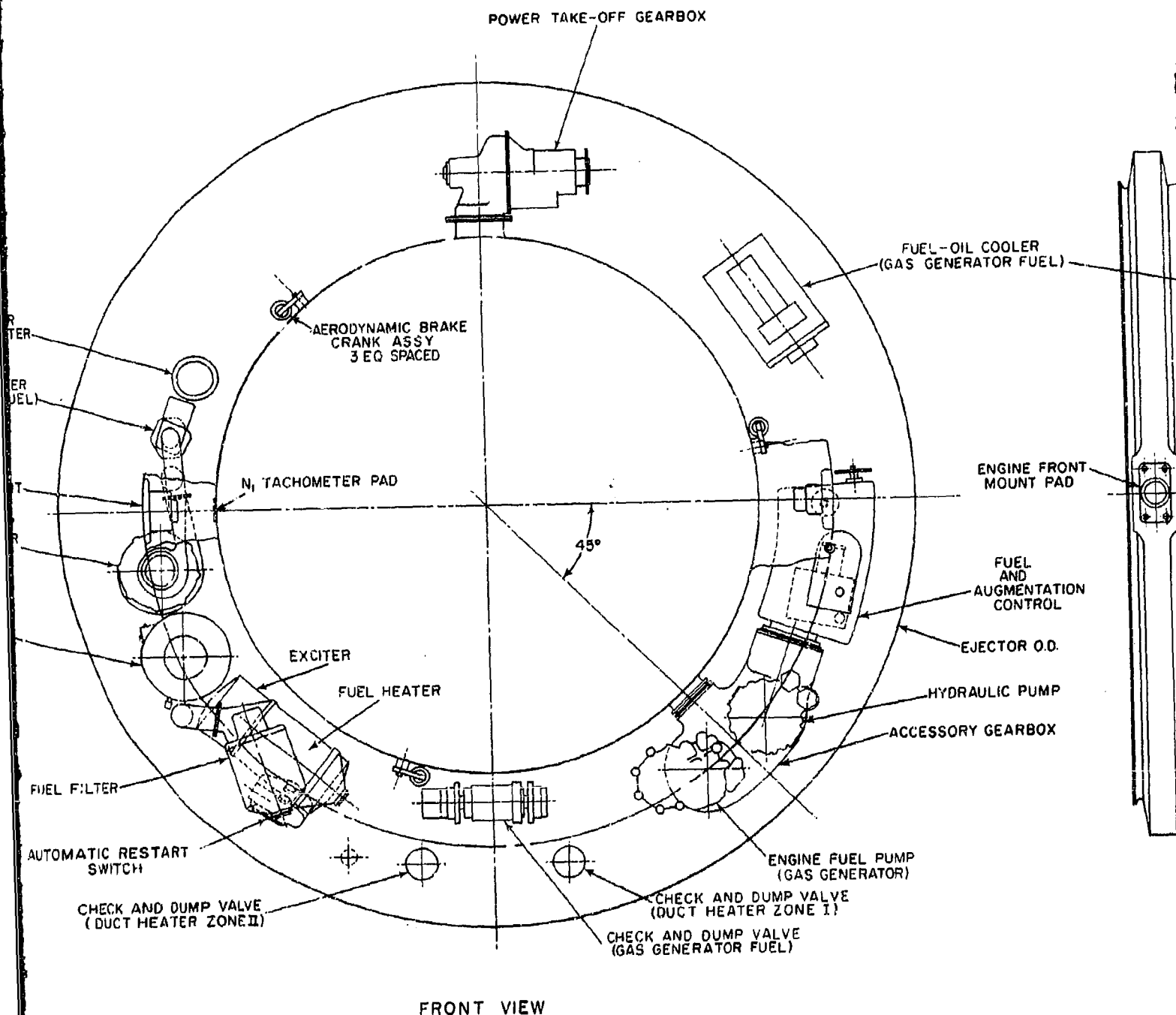




Figure 1-28

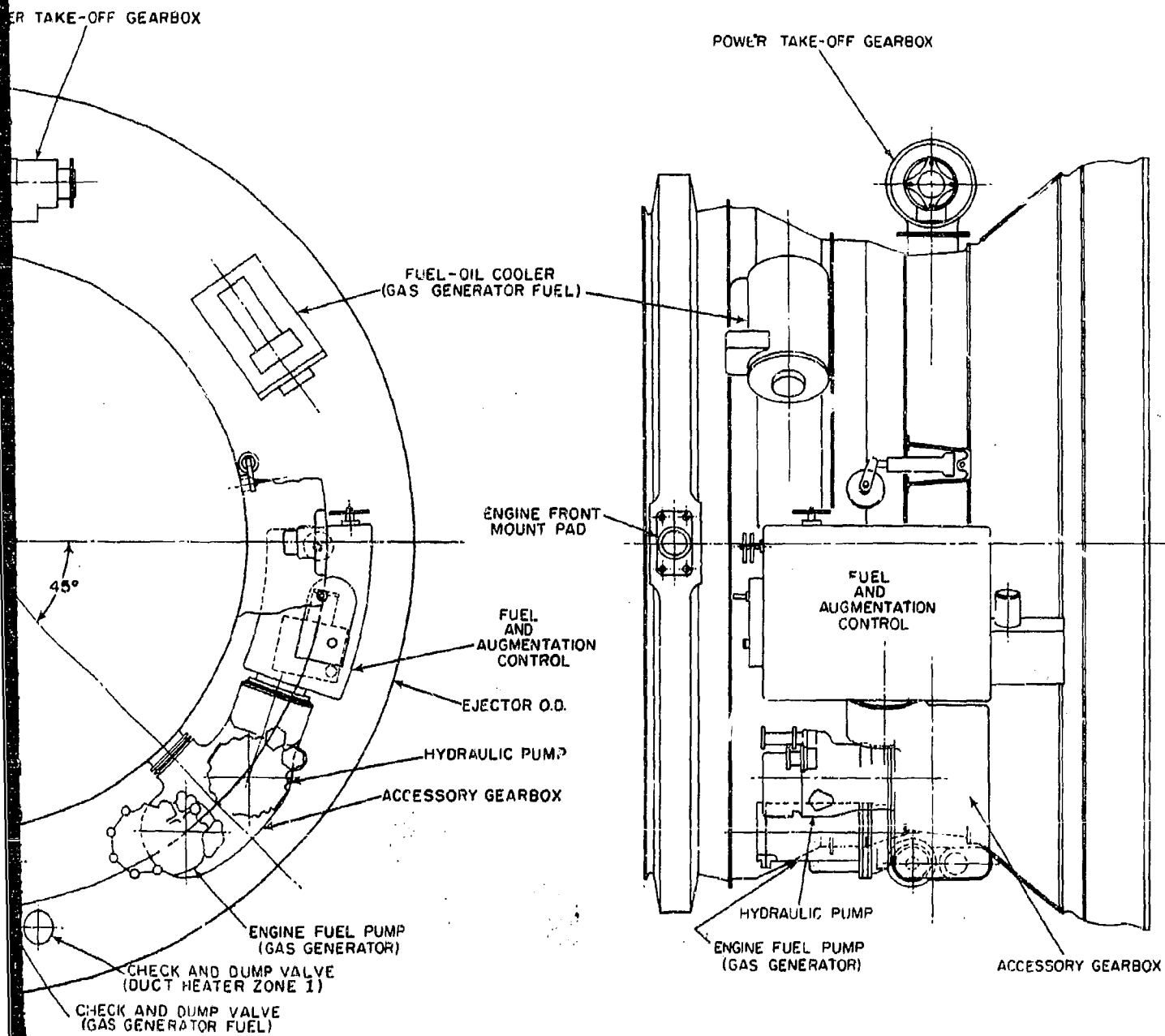
[illegible]





REVISED ACCESSORY ARRANGEMENT  
MOUNT STRUCTURE AT

Figure 1-20



LEFT SIDE VIEW

REVISED ACCESSORY ARRANGEMENT MAKING ROOM FOR  
MOUNT STRUCTURE AT TOP OF ENGINE

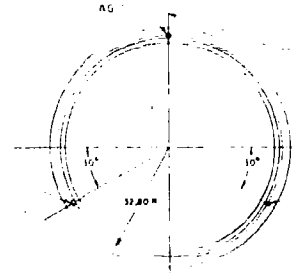
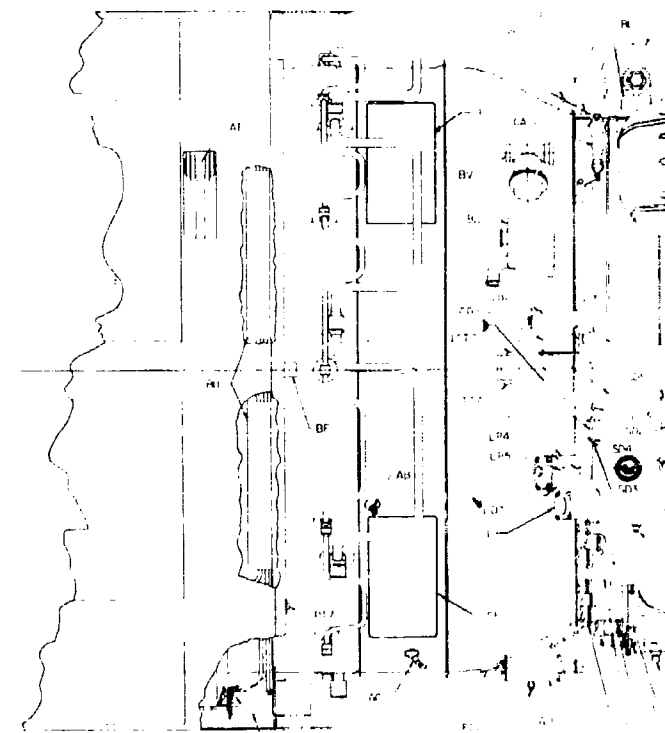
Figure 1-29

EXAMINATED AT 5 YEAR INTERVALS  
REPLACEMENT AFTER 10 YEARS  
1000 (M 884) 10

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3

ACCESSORY DRIVE PADS			
100-100	G	FUEL TANK FILL	100-100
100-100	H	ACCESSORY DRIVE PADS	100-100
100-100	I	ACCESSORY DRIVE PADS	100-100
FUEL DRAIN			
100-100	FD1	CONSUMPTION CHAMBER FUEL DRAIN	100-100
100-100	FD2	CLAMP VALVE DRAIN GAS (GEN)	100-100
100-100	FD3	CLAMP VALVE DRAIN GAS (GEN)	100-100
100-100	FD4	CLAMP VALVE DRAIN GAS (GEN)	100-100
FUEL PRESSURE			
100-100	FD1	FUEL PUMP INLET PRESSURE	100-100
100-100	FD3	CLAMP VALVE FUEL PRESSURE	100-100
FUEL FLOW			
100-100	F1	FUEL PUMP SUPPLY INLET	100-100
100-100	F2	MAIN FUEL FLOWMETER SUPPLY INLET	100-100
100-100	F3	MAIN FUEL FLOWMETER SUPPLY OUTLET	100-100
100-100	F4	DUCT HEATER FUEL PUMP VALVE	100-100
100-100	F5	DUCT HEATER FLOWMETER SUPPLY INLET	100-100
100-100	F6	DUCT HEATER FLOWMETER SUPPLY OUTLET	100-100
FUEL VENT			
100-100	FV1	FUEL PUMP OUTLET VENT	100-100
FUEL TEMPERATURE			
100-100	FT1	HEATER OUTLET FUEL TEMP	100-100
OIL BEZATHEE			
100-100	OT1	MAIN OIL INCHORDING BEZATHEE	100-100
OIL DRAIN			
100-100	OD1	OIL TANK DRAIN	100-100
100-100	OD2	OIL TANK OVERFLOW DRAIN	100-100
100-100	OD3	OIL TANK OVERFLOW DRAIN	100-100
100-100	OD4	OIL TANK OVERFLOW DRAIN	100-100
100-100	OD5	OIL TANK OVERFLOW DRAIN	100-100
OIL FLOW			
100-100	OF1	OIL TANK REMOTE FILLER	100-100
100-100	OF2	OIL TANK MANUAL FILL	100-100
OIL PRESSURE			
100-100	OP1	PRESSURE FOR TANKING TIE	100-100
100-100	OP4	OIL FILTER INLET PRESSURE (PROV FOR AP)	100-100
100-100	OP5	OIL FILTER OUTLET PRESSURE	100-100
OIL TEMPERATURE			
100-100	OT1	MAIN OIL TEMP	100-100
OIL VENT			
100-100	OV1	OIL PRESSURE TRANSMITTER VENT	100-100
SEAL DRAIN			
100-100	SD1	FUEL CONTROL SEAL DRAIN	100-100
100-100	SD2	FUEL PUMP SEAL DRAIN	100-100
100-100	SD3	HYDRAULIC PUMP SEAL DRAIN	100-100
100-100	SD4	DUCT HEATER FUEL PUMP SEAL DRAIN	100-100
100-100	SD5	ACCESSORY DRIVE OVERBOARD DRAIN (TACHOMETER)	100-100
100-100	SD6	ACCESSORY DRIVE OVERBOARD DRAIN (INTO PUMP)	100-100
100-100	SD7	ACCESSORY DRIVE OVERBOARD DRAIN (FUEL PUMP)	100-100
TEMPERATURE SENSING			
100-100	TS1	TURBINE EXIT TEMP (AUG)	100-100
100-100	TS2	TURBINE EXIT TEMP (INDIVIDUAL)	100-100
PRESSURE SENSING			
100-100	PT1	TURBINE EXIT PRESSURE	100-100
MISCELLANEOUS			
100-100	M1	OIL AIR	100-100
100-100	M2	GAS GEN FUEL PUMP	100-100
100-100	M3	FUEL PUMP FILTER (GEN)	100-100
100-100	M4	FUEL PUMP FILTER (GEN) (FOR REMOVAL)	100-100
100-100	M5	DUCT HEATER FUEL PUMP	100-100
100-100	M6	HYDRAULIC PUMP	100-100
100-100	M7	OIL PUMP	100-100
100-100	M8	FUEL OIL COOLER (GAS GEN)	100-100
100-100	M9	FUEL OIL COOLER (DUCT HEATER)	100-100
100-100	M10	GEARBOX	100-100
100-100	M11	AUTOMATIC RESTART SWITCH	100-100
100-100	M12	BREATHING PRESSURIZATION VALVE	100-100
100-100	M13	HIGH PRESSURE BLEED PAD	100-100
100-100	M14	UNLIMITED CONTROL (FUEL, AUGMENTATION, & NOZZLES)	100-100
100-100	M15	DUCT TURBOPUMP CONTROLLER	100-100
100-100	M16	EXHAUST FUEL (GAS GEN)	100-100
100-100	M17	WATER FUEL (DUCT HEATER)	100-100
100-100	M18	ENGINE PROPT MOUNTING PROVISIONS	100-100
100-100	M19	ENGINE REAR MOUNTING PROVISIONS	100-100
100-100	M20	FUEL INJECTION TO SUPPLEMENTARY COOLING	100-100
100-100	M21	GROUND HANDLING HOLES (PROPT MOUNT)	100-100
100-100	M22	GROUND HANDLING AREA (REAR)	100-100
100-100	M23	POWER CONTROL, SHUT OFF ANGLE OF TRAVEL	100-100
100-100	M24	SHUT OFF ENGINE (SHUT ANGLE OF TRAVEL)	100-100
100-100	M25	APPROACH VELOCITY CONTROL (SHUT ANGLE OF TRAVEL)	100-100
100-100	M26	WASHING ON SHUT POSITION (SHUT ANGLE OF TRAVEL)	100-100
100-100	M27	AIR TURBINE VALVE (DUCT HEATER FUEL PUMP)	100-100
100-100	M28	IGNITION EXHAUST ELECTRICAL CONT. (GAS GEN)	100-100
100-100	M29	THERMAL AUTO REGR. PAD	100-100
100-100	M30	AUTOMATIC BRAKE CONTROL AIR SUPPLY CONT.	100-100
100-100	M31	DUCT NOZZLE FEEDBACK	100-100
100-100	M32	DUCT NOZZLE FEEDBACK (GEN)	100-100
100-100	M33	GAS GEN FUEL FLOWMETER (GEN)	100-100
100-100	M34	GAS GEN FUEL FLOWMETER (GEN)	100-100
100-100	M35	DUCT HEATER FUEL FLOWMETER (GEN)	100-100
100-100	M36	FUEL CONTROL FUEL FILTER	100-100
100-100	M37	CHECK & DUMP VALVE (GEN)	100-100
100-100	M38	CHECK & DUMP VALVE FILTER	100-100
100-100	M39	NOZZLE POSITION INDICATOR MOUNTING PROVISIONS	100-100
100-100	M40	REVERSE POSITION INDICATOR CONT. (ELECTRICAL)	100-100
100-100	M41	HYDRAULIC FUEL BYPASS VALVE	100-100
100-100	M42	IGNITION EXHAUST ELECTRICAL CONT. (DUCT HEATER)	100-100
100-100	M43	POWER TAKEOFF GEARBOX	100-100
100-100	M44	FUEL HEATER VALVE CONT. (ELECTRICAL)	100-100
100-100	M45	FUEL HEATER VALVE POSITION INDICATOR (ELECTRICAL)	100-100
100-100	M46	POWER TAKEOFF DECOUPLER	100-100

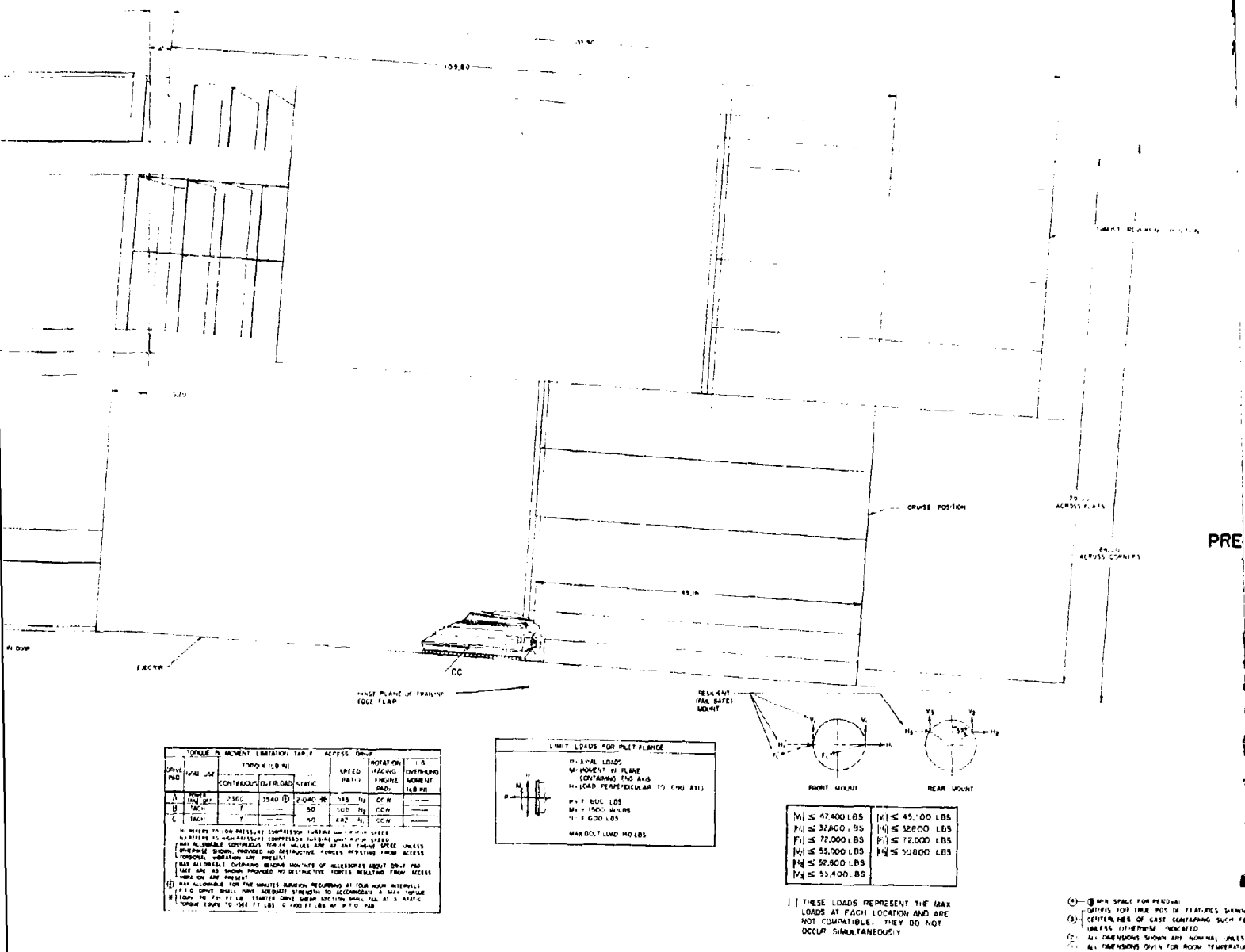


FRONT MOUNT RING  
VIEWED FROM INLET END  
SCALE 1:1









PROPOSED TURBOFAN ACCESSORY ARRANG

Figure 1-30



PRELIMINARY INSTALLATION DWG  
PRINTED JUNE 23, 1965

**LIMIT LOADS FOR INLET FLANGE**

P-AXIAL LOADS  
MOMENT IN PLANE  
CONTAINING ENG AXIS  
M-HEAD PERPENDICULAR TO ENG AXIS

P-AXIAL LOADS  
MAX 1500 LBS  
MIN 0 LBS

M-HEAD LOADS  
MAX 1500 LBS  
MIN 0 LBS

$N_1 \leq 47,400 \text{ LBS}$	$N_2 \leq 45,100 \text{ LBS}$
$N_3 \leq 32,800 \text{ LBS}$	$N_4 \leq 32,800 \text{ LBS}$
$N_5 \leq 72,000 \text{ LBS}$	$N_6 \leq 72,000 \text{ LBS}$
$N_7 \leq 55,000 \text{ LBS}$	$N_8 \leq 50,800 \text{ LBS}$
$N_9 \leq 52,800 \text{ LBS}$	$N_{10} \leq 55,400 \text{ LBS}$

THESE LOADS REPRESENT THE MAX  
LOADS AT EACH LOCATION AND ARE  
NOT COMPATIBLE. THEY DO NOT  
OCCUR SIMULTANEOUSLY

- MIN SPACE FOR REMOVAL
- DATA FOR TIME POS OF FEATURES SHOWN ARE  
CHARACTERISTICS OF CASE CONTAINING SUCH FEATURES  
UNLESS OTHERWISE INDICATED
- ALL DIMENSIONS SHOWN ARE NOMINAL UNLESS OTHERWISE SPECIFIED
- ALL DIMENSIONS GIVEN FOR ROOM TEMPERATURE UNLESS OTHERWISE NOTED

## PROPOSED TURBOFAN ACCESSORY ARRANGEMENT

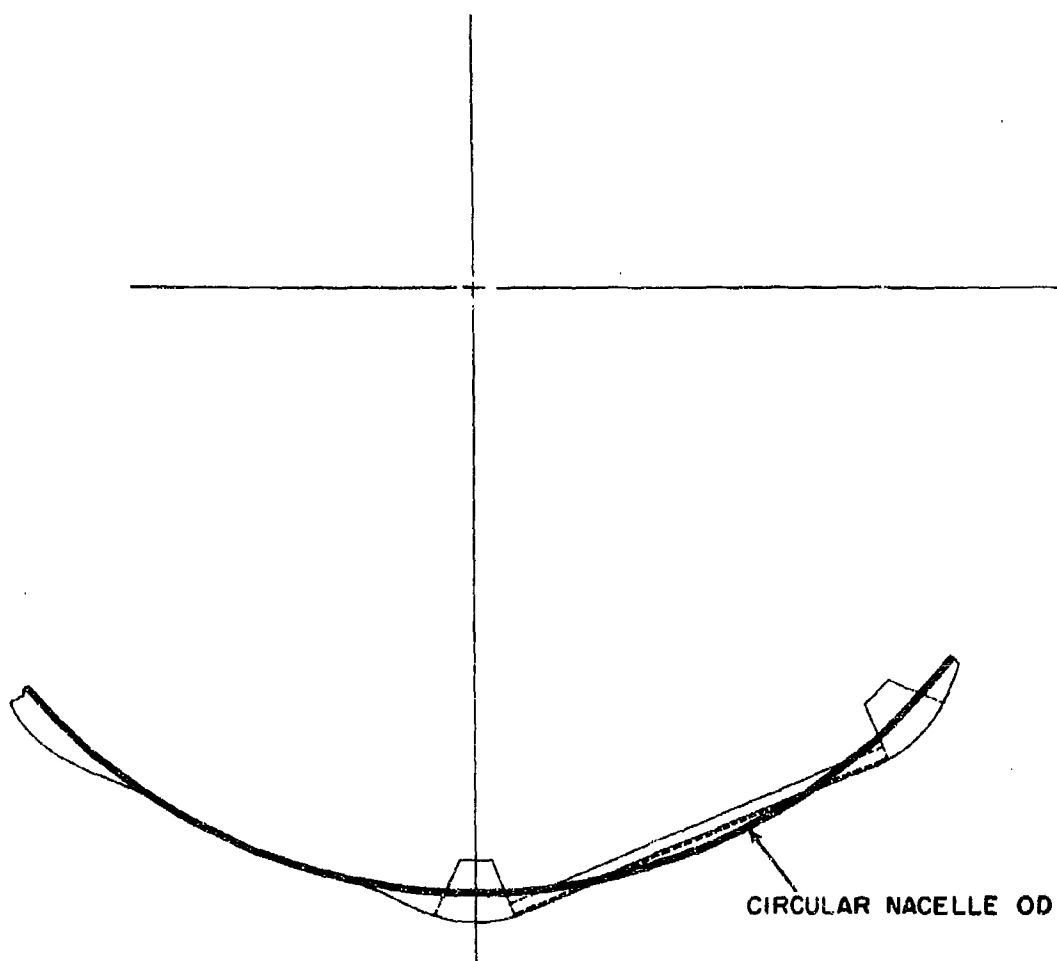
Figure 1-30

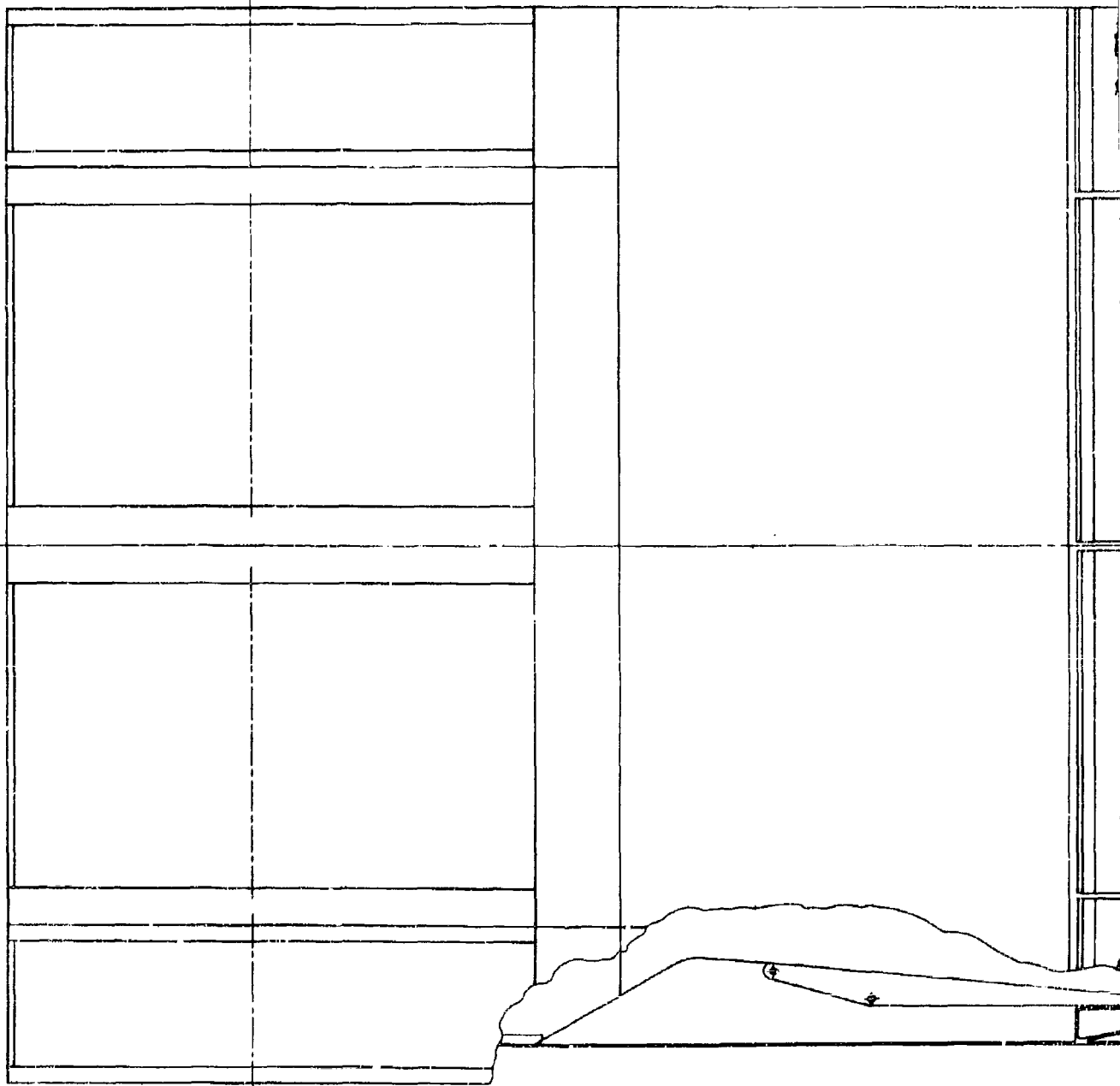
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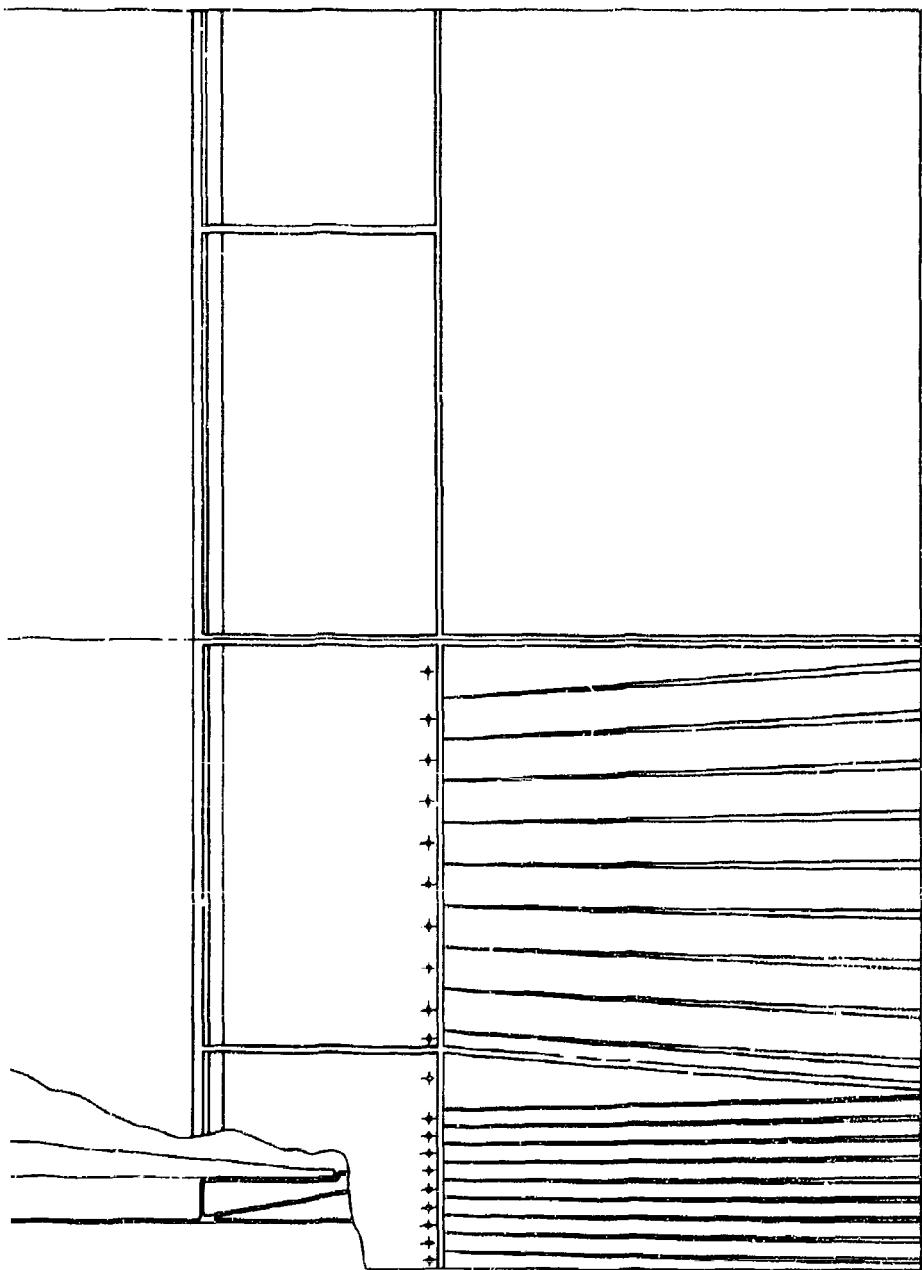
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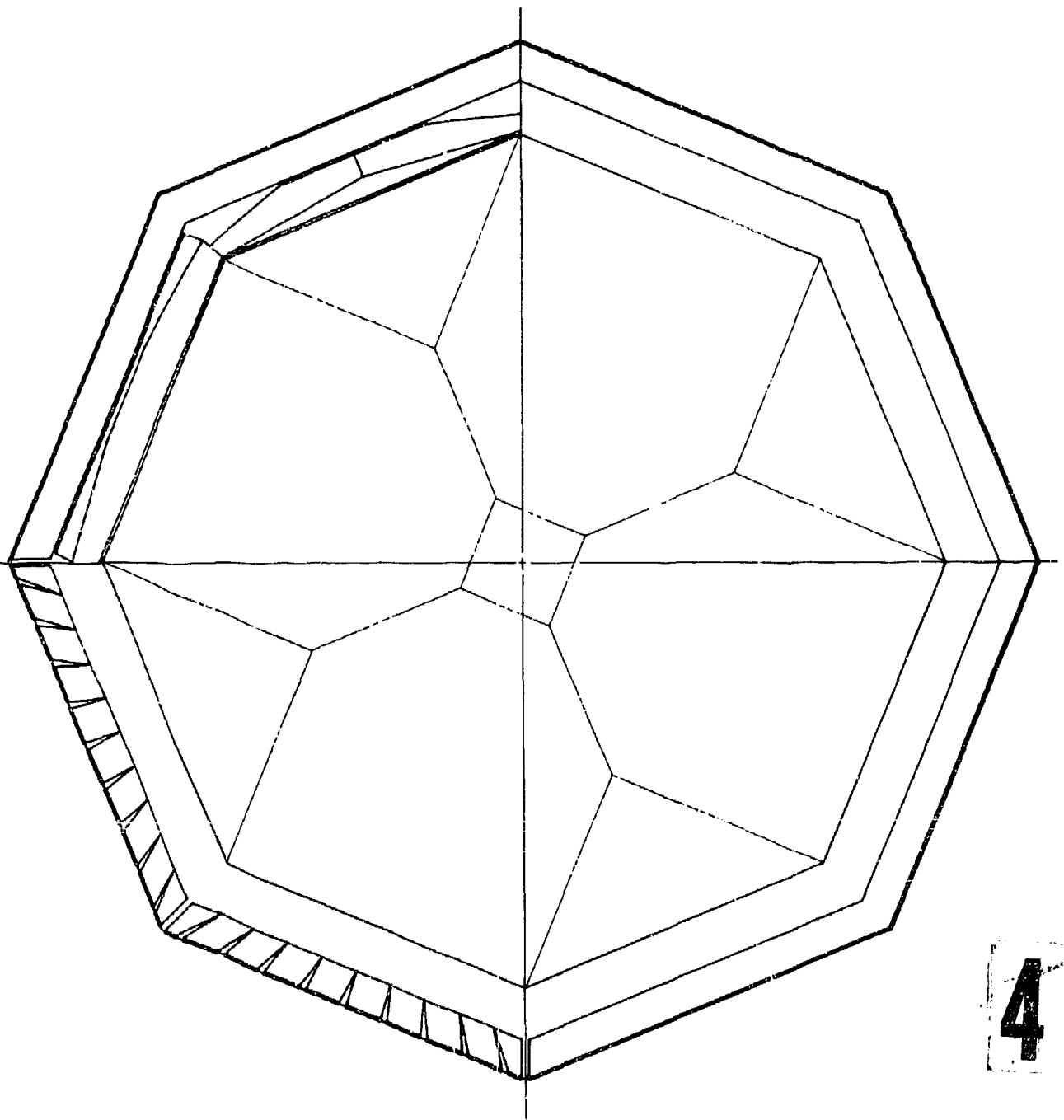






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SCHEMATIC OF OCTAGONAL EJECTOR

Figure 1-31

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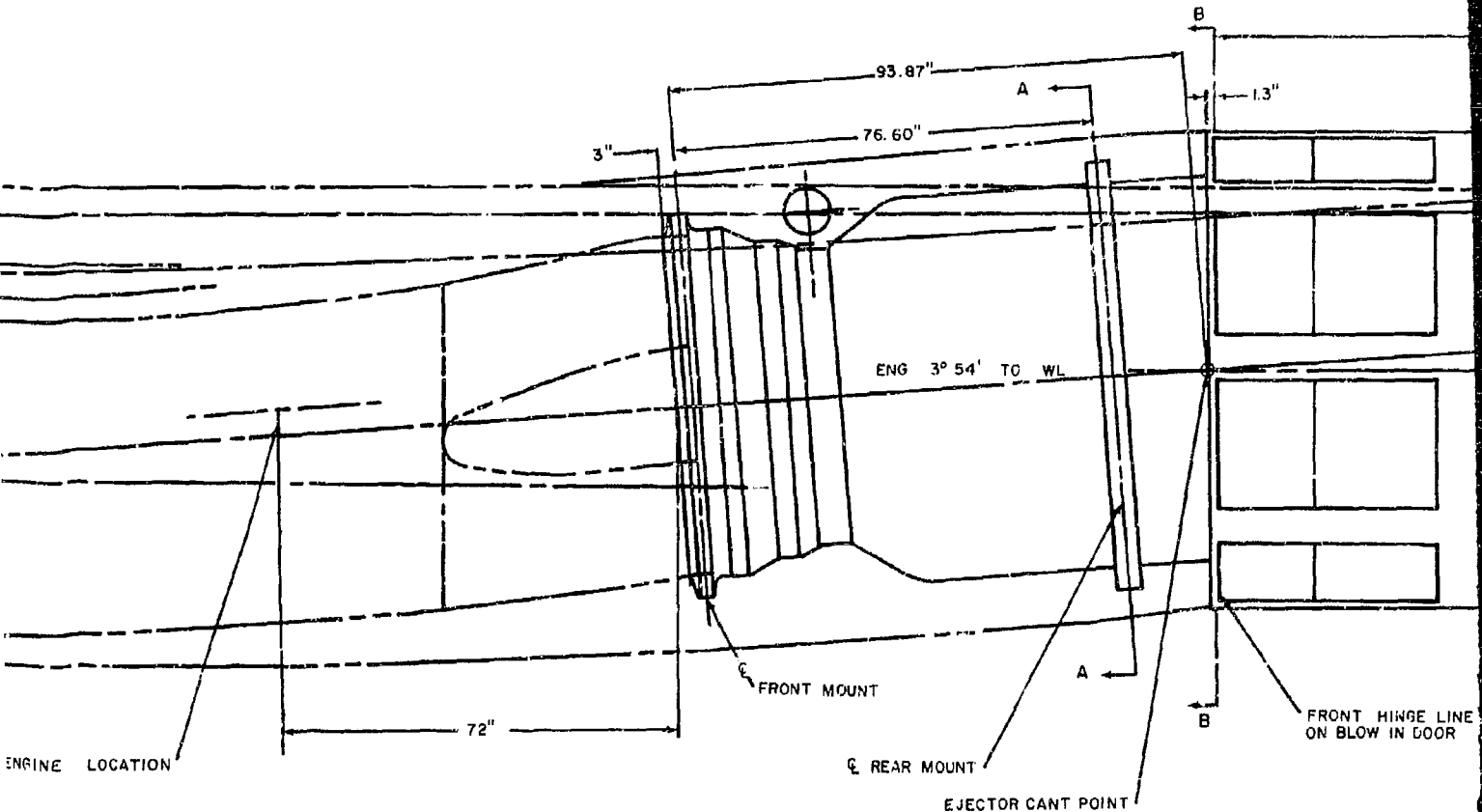
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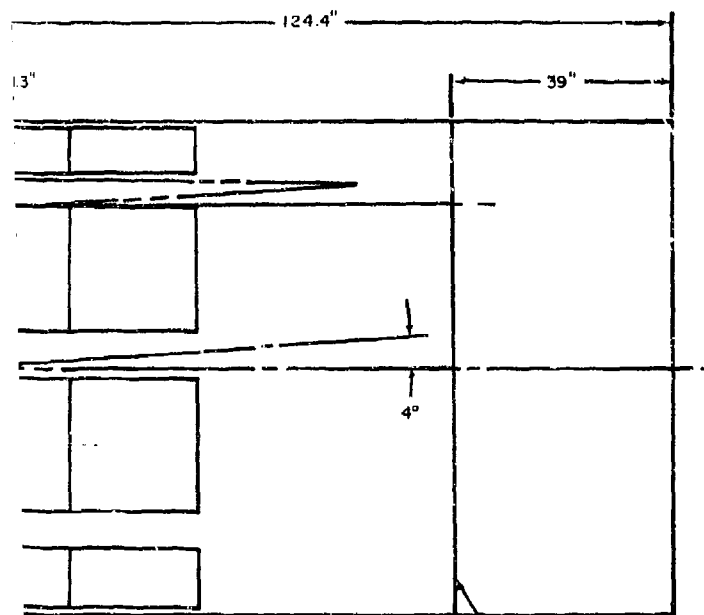
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WRP

PRESENT ENGINE LOCATION



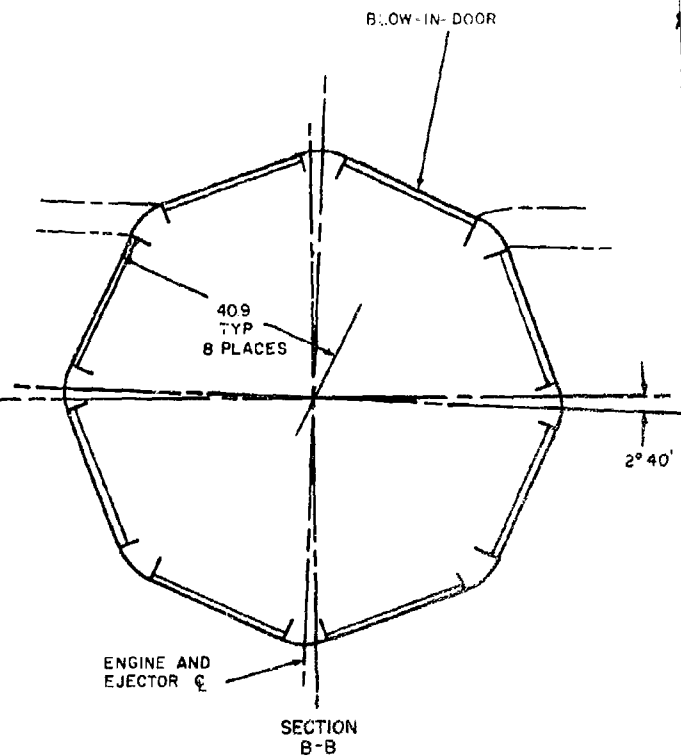




FRONT HINGE LINE  
ON BLOW IN DOOR

FRONT HINGE LINE  
ON TAIL FEATHERS

EJECTOR CANTED AT  
REAR MOUNT PLANE

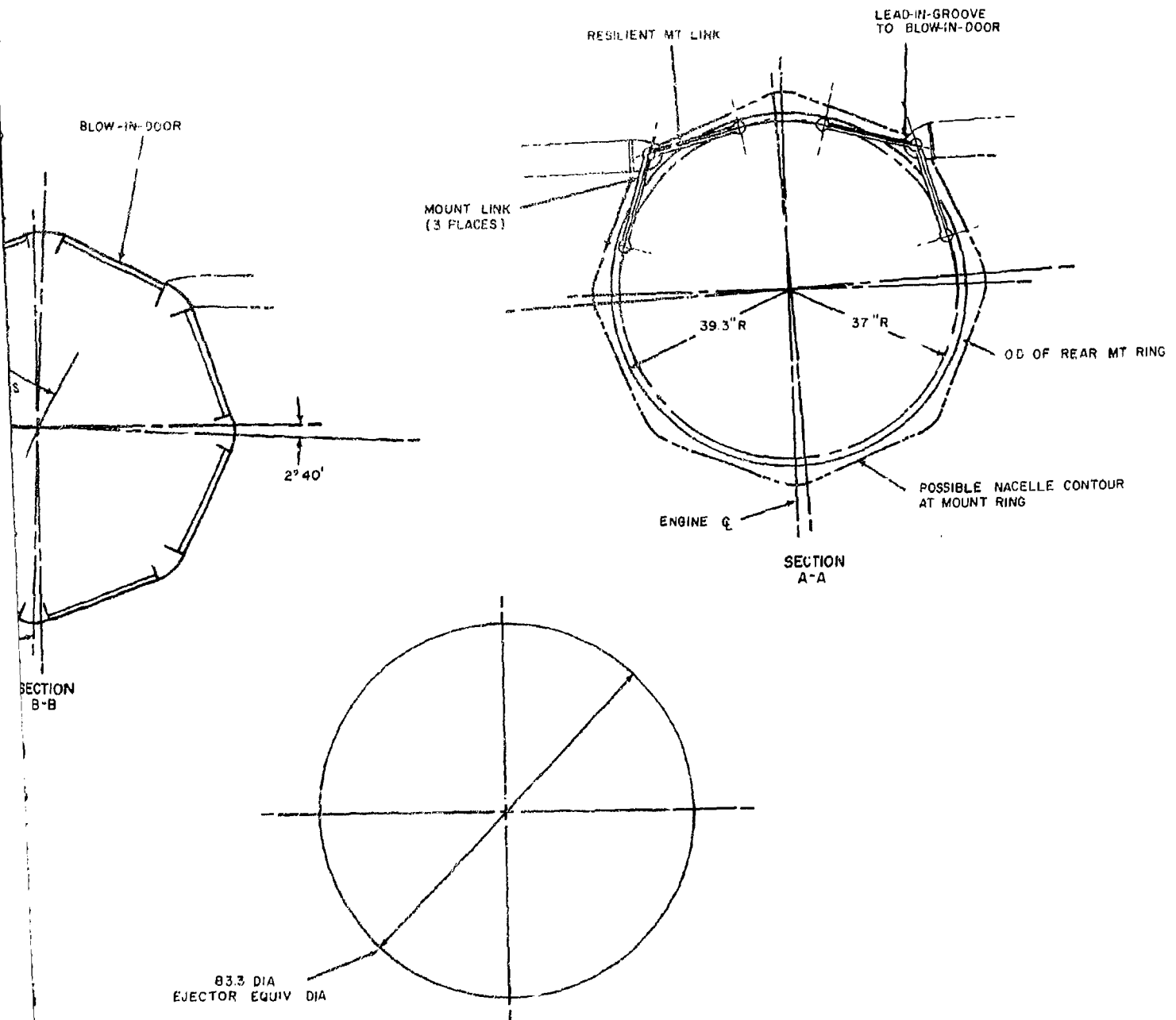


83.3 DIA  
EJECTOR EQUIV DIA

3

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STF219L 700 LB/SEC TURBOFAN OUTBOARD ENGINE

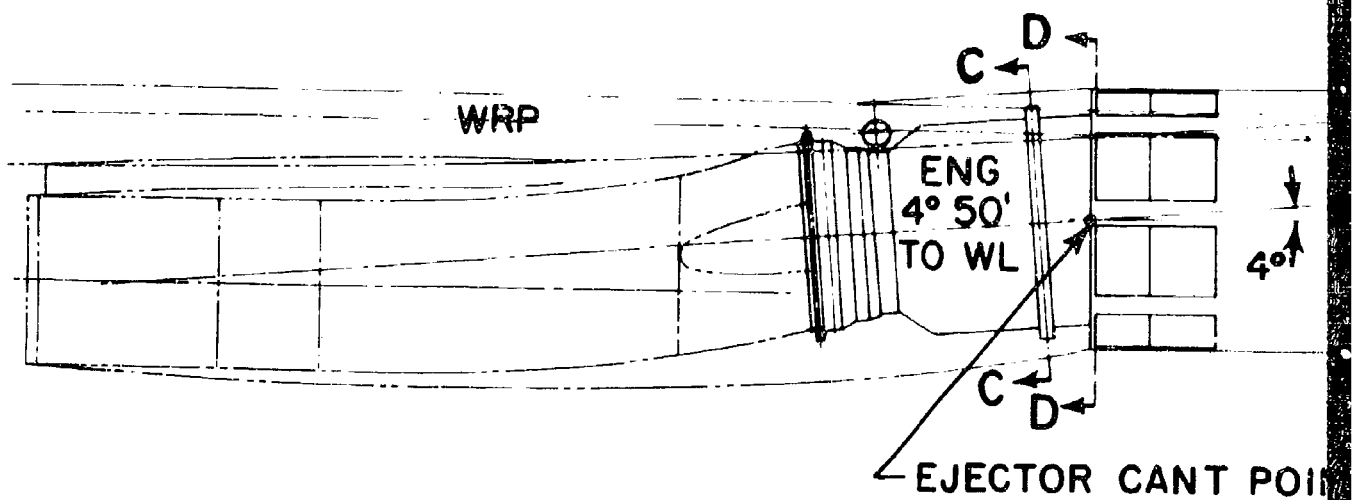
Figure 1-32

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LINE UP BLOW-IN-DOORS  
WITH WING LIKE THIS

SECTION C-C

ENGINE AND  
EJECTOR  $\phi$

SECTION D-D

83.3 DIA  
EJECTOR EQUIV DIA

39.3" R

37" R

ENGINE  $\phi$

POSSIBLE  
NACELLE  
CONTOUR  
AT  
MOUNT  
RING

POINT

2

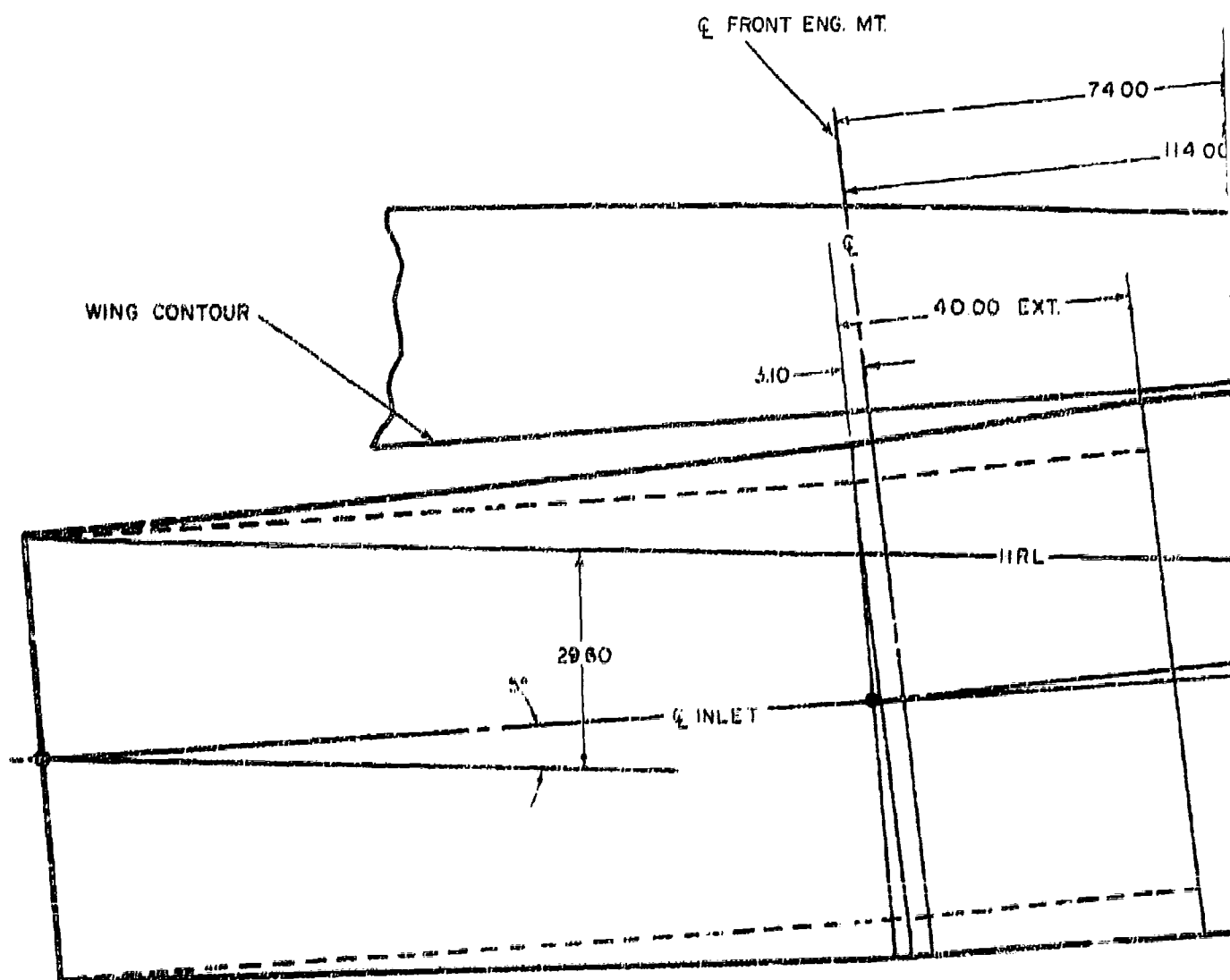
STF219L 700 L.B/SEC TURBOFAN INBOARD ENGINE

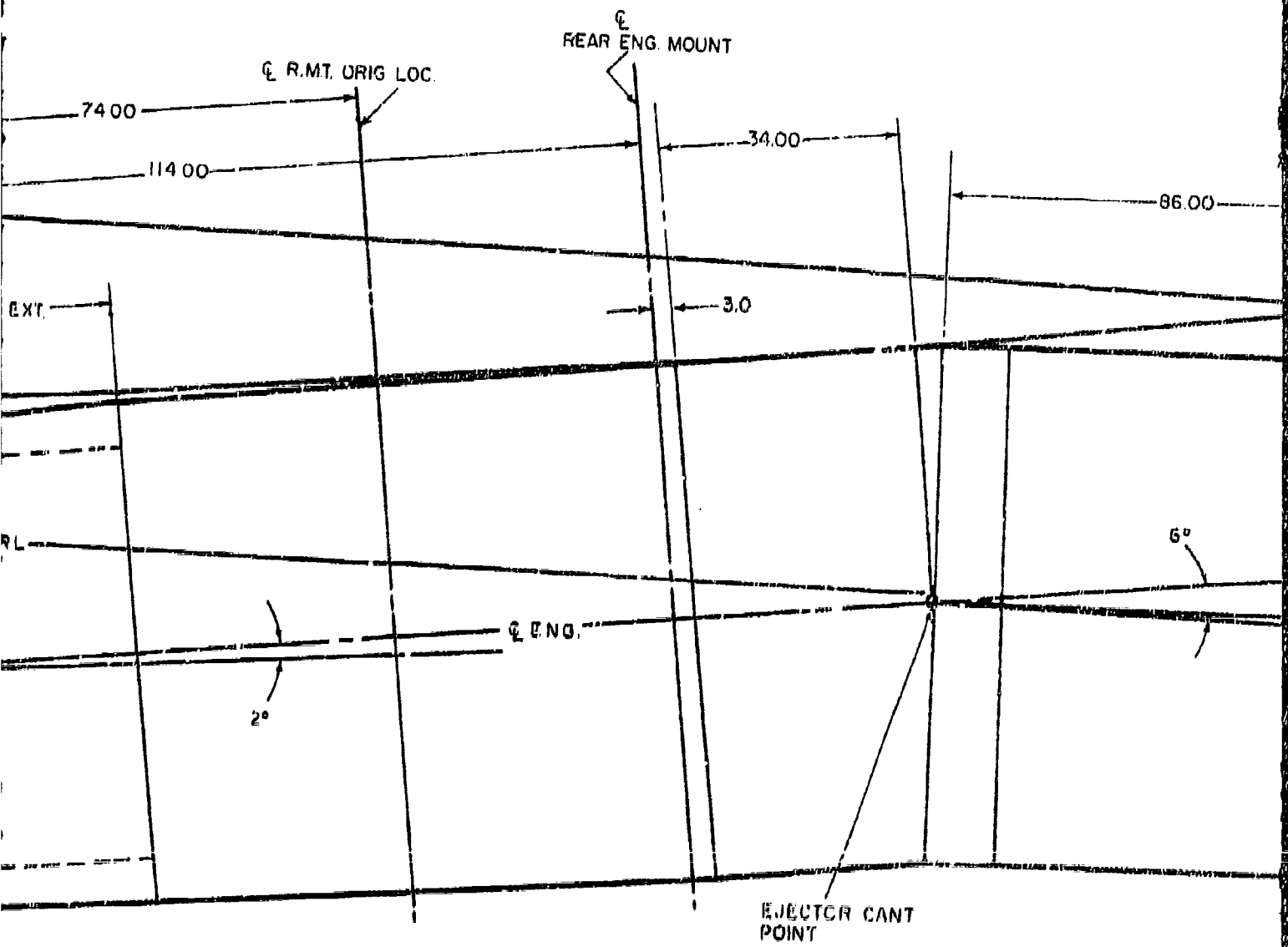
Figure 1-33

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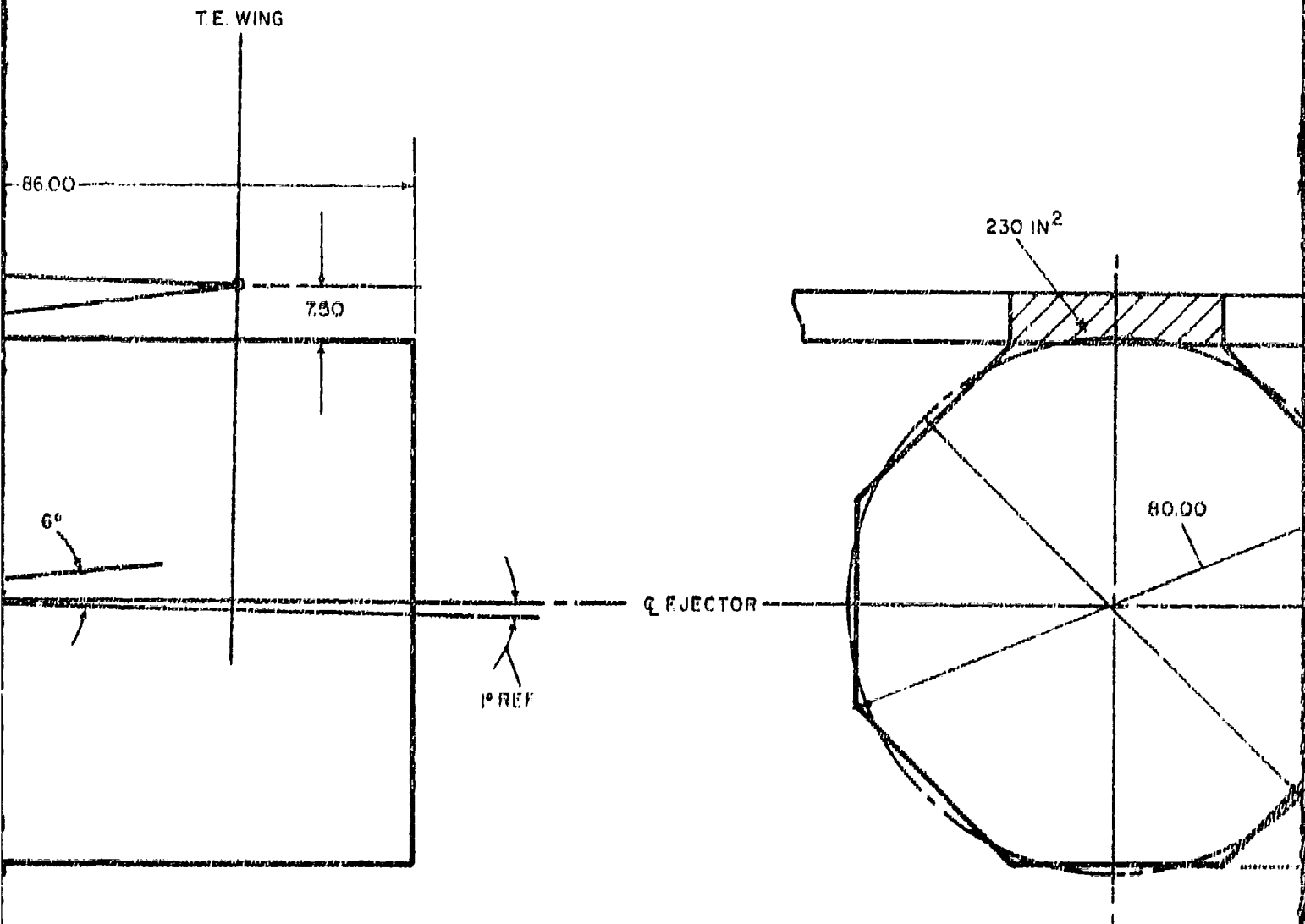
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PRATT & WHITNEY AIRCRAFT





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ARRANGEMENT SHOWING REAR ENGL  
MOVED AFT 40.00  
ENGINE & EJECTOR ARE TANGENT TO  
EJECTOR CANTED AT NOZZLE PLANE  
ENGL MOUNT PLANES ARE PARALLEL  
BASE DRAG AREA 230 in<sup>2</sup>

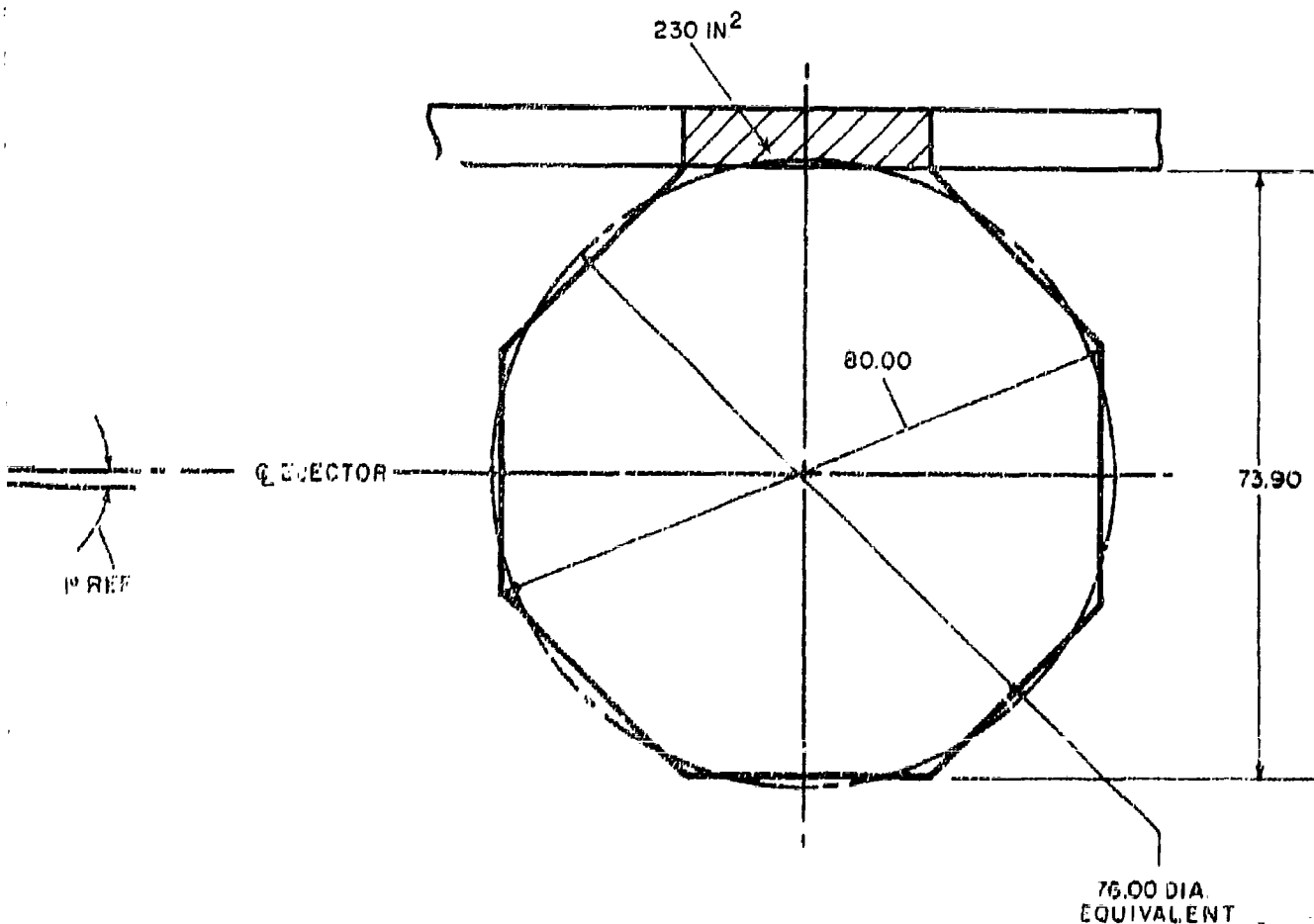
STP219L 650 LB/SEC TURBOFAN

Figure 1-14

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ARRANGEMENT SHOWING REAR ENCL. MOUNT  
MOVED AFT. 40.00  
ENGINE & EJECTOR ARE TANGENT TO WING  
EJECTOR CANTED AT NOZZLE PLANE  
ENG. MOUNT PLANES ARE PARALLEL  
BASE DRAG AREA 230 in<sup>2</sup>

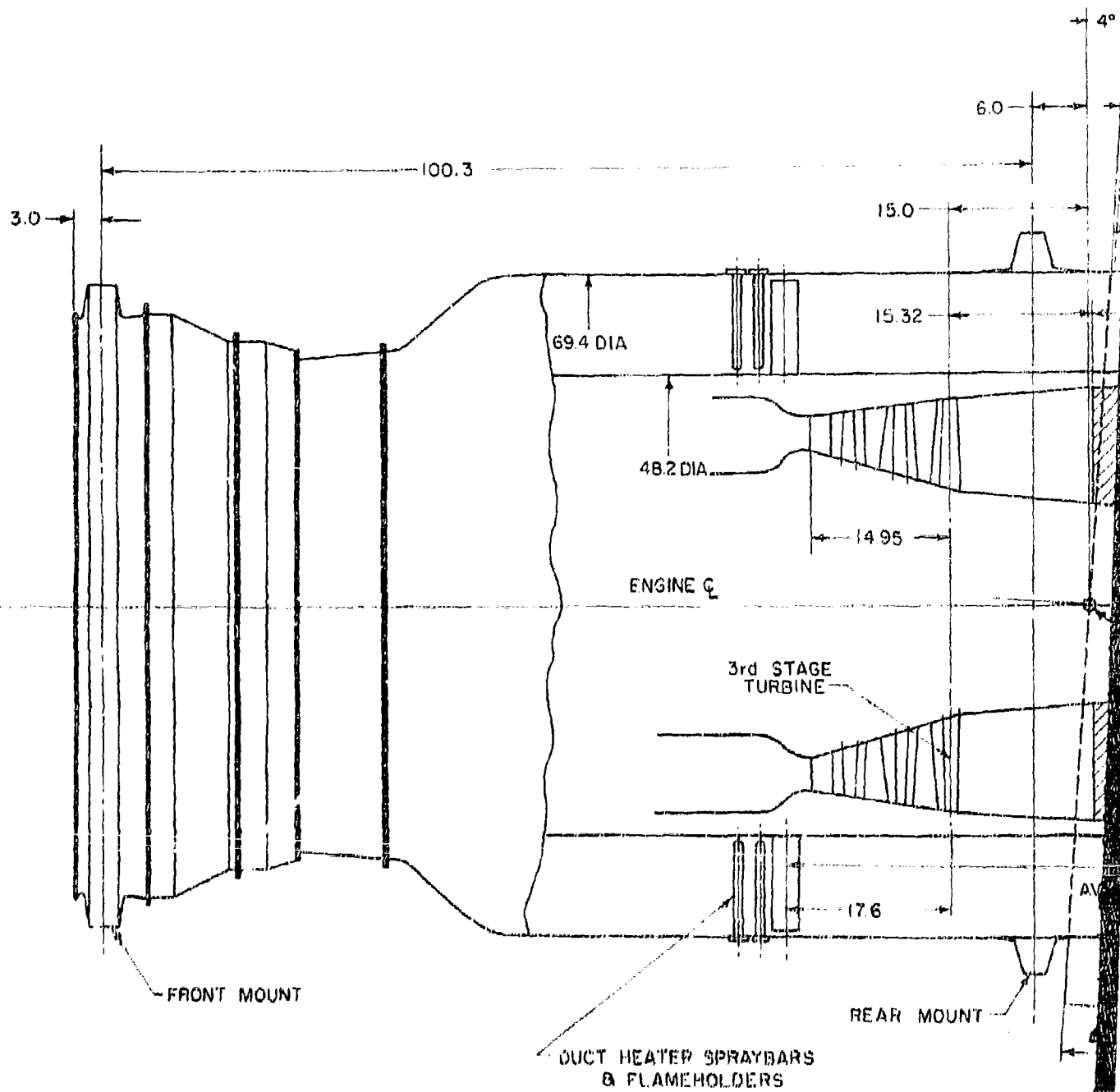
STE219L 650 LB/SEC TURBOFAN

Figure 1-34

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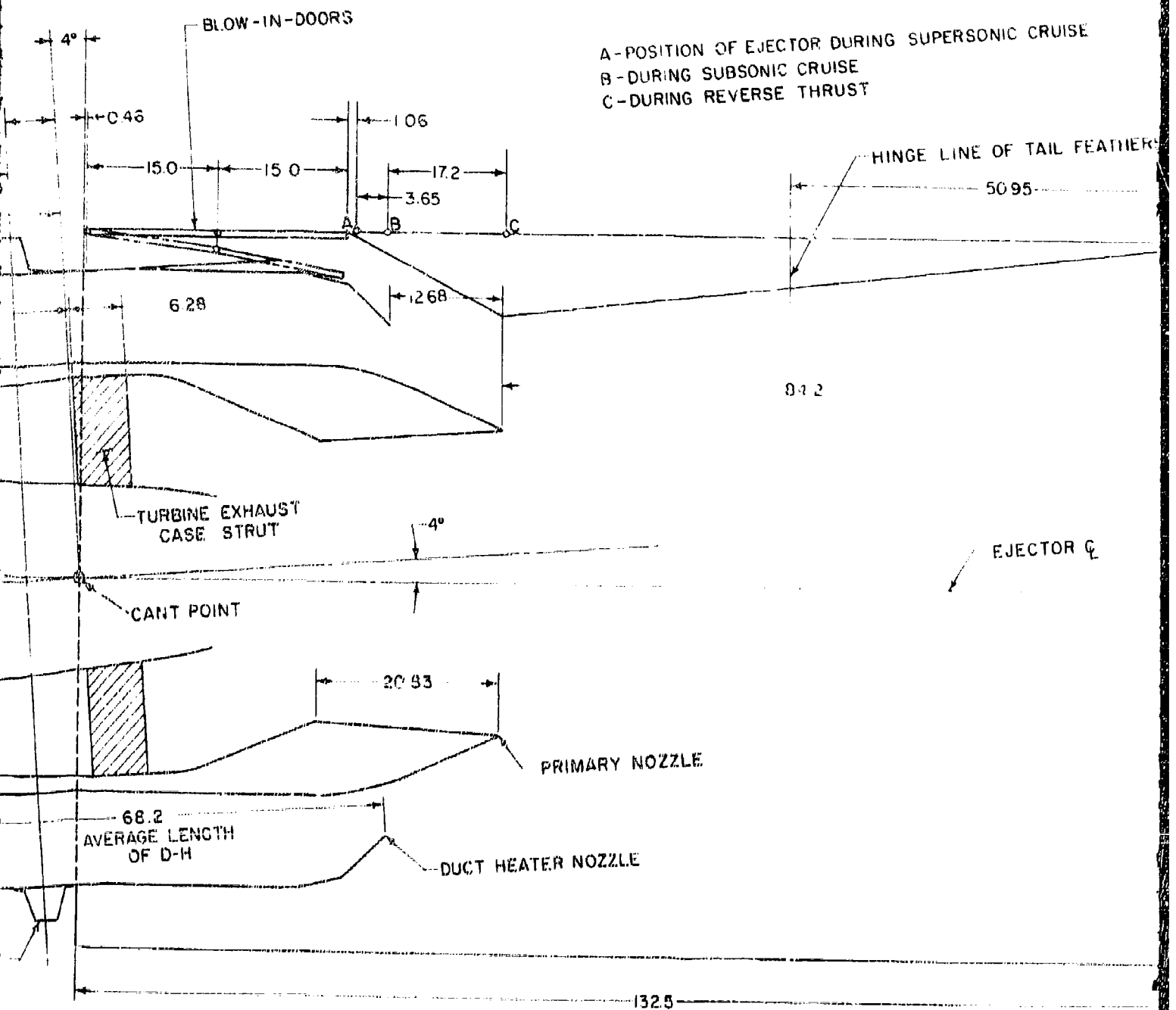
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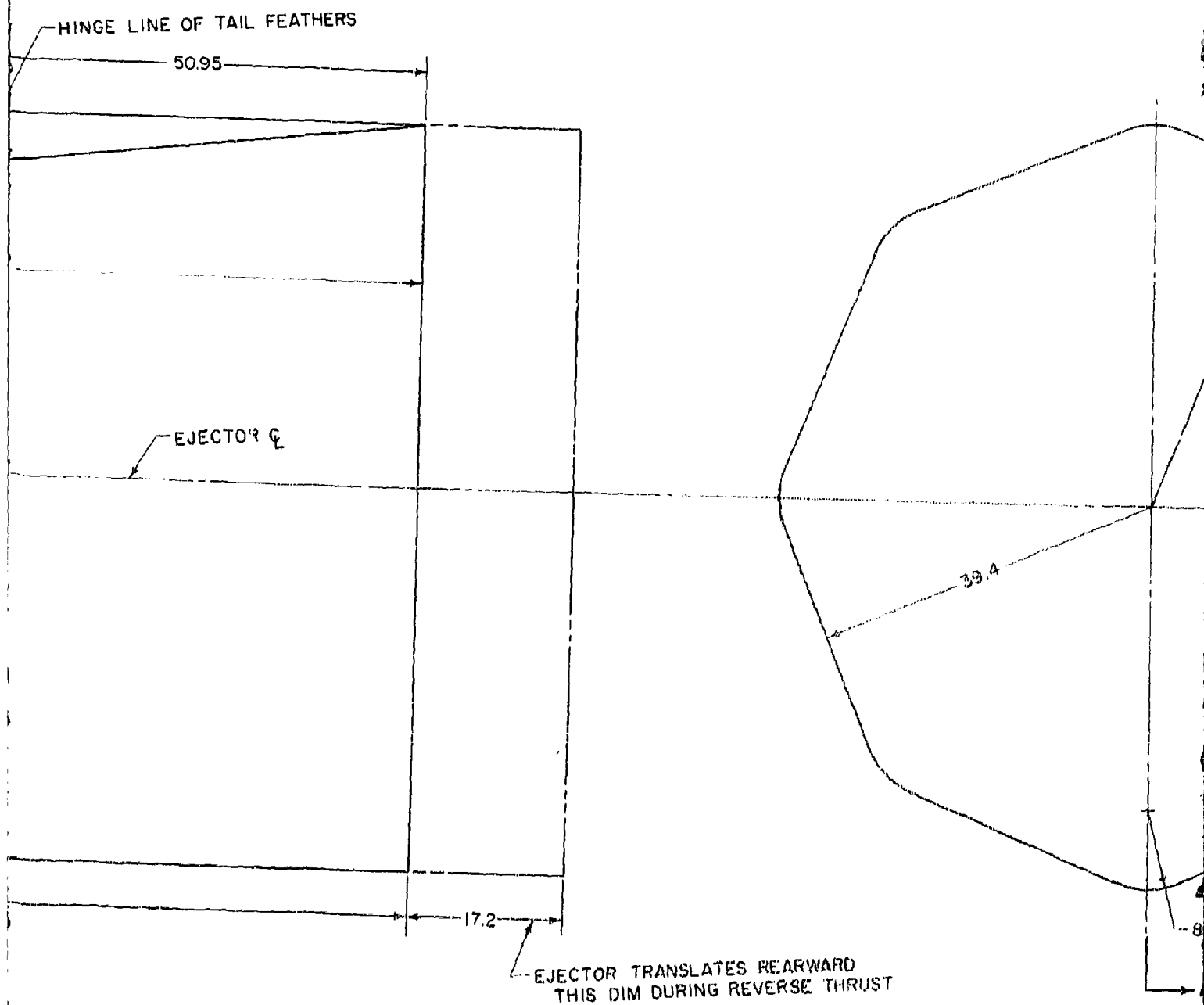




SECTION A-A

DURING SUPERSONIC CRUISE  
CRUISE  
CRUISE

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STF219L 650 LB/SEC TURBOFAN  
ENGINE/EJECTOR RELATION

Figure 1-35

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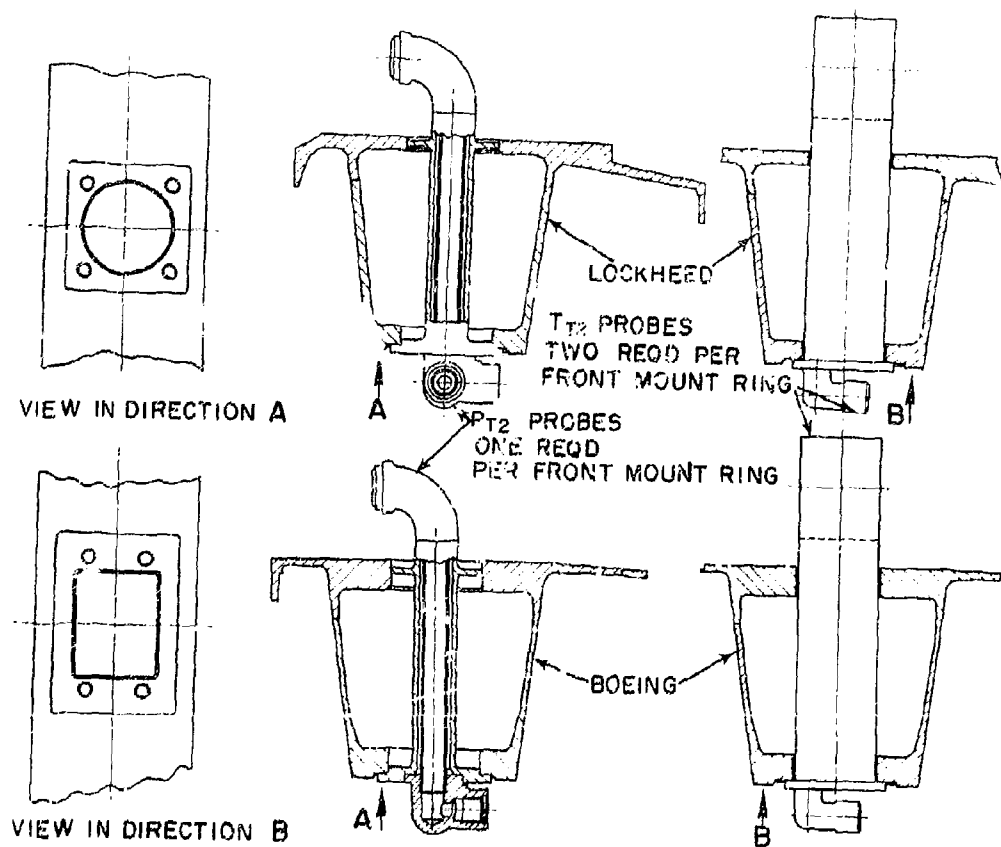
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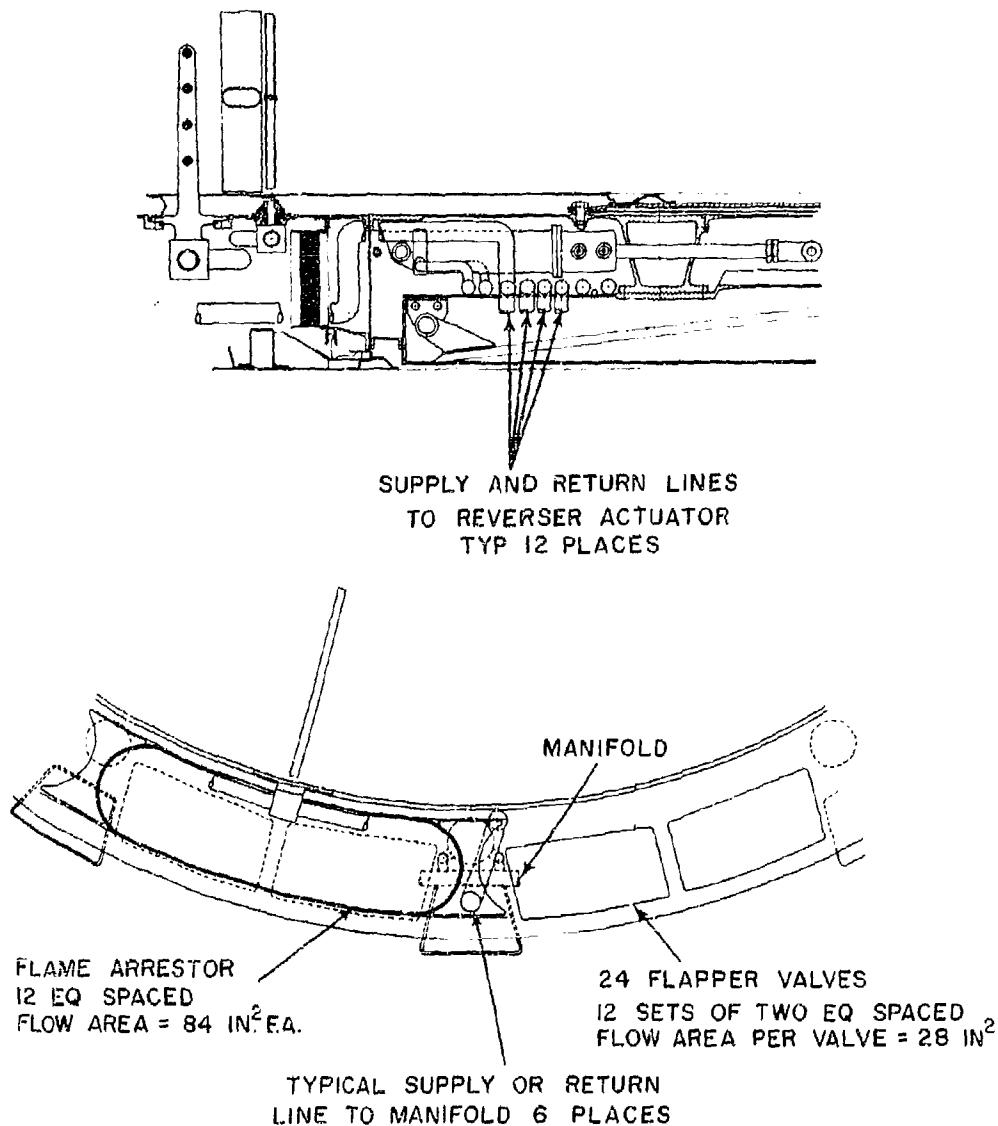
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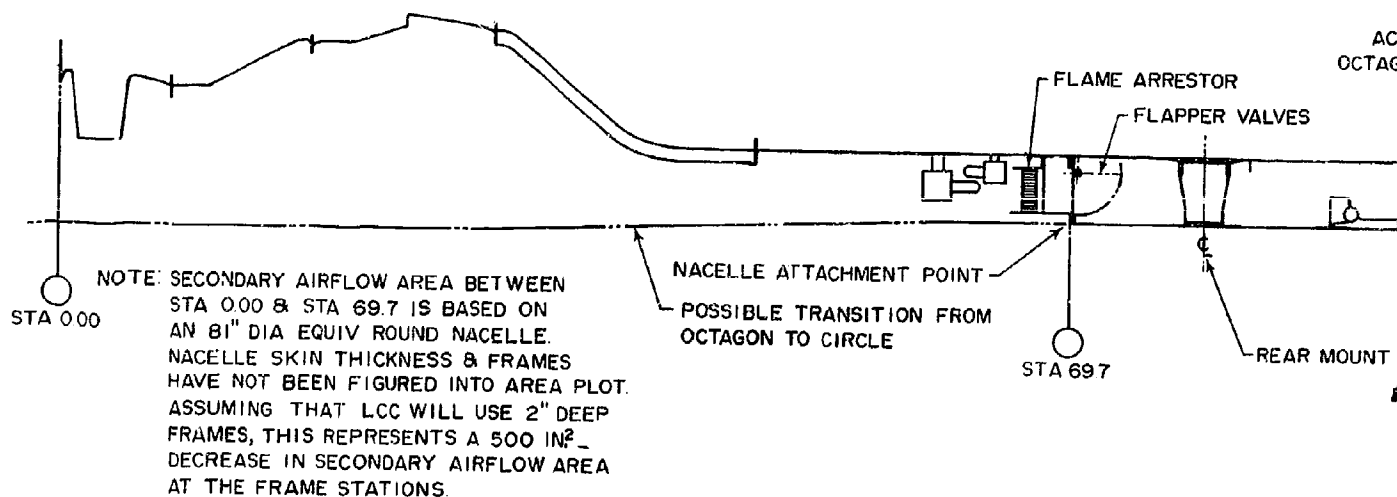
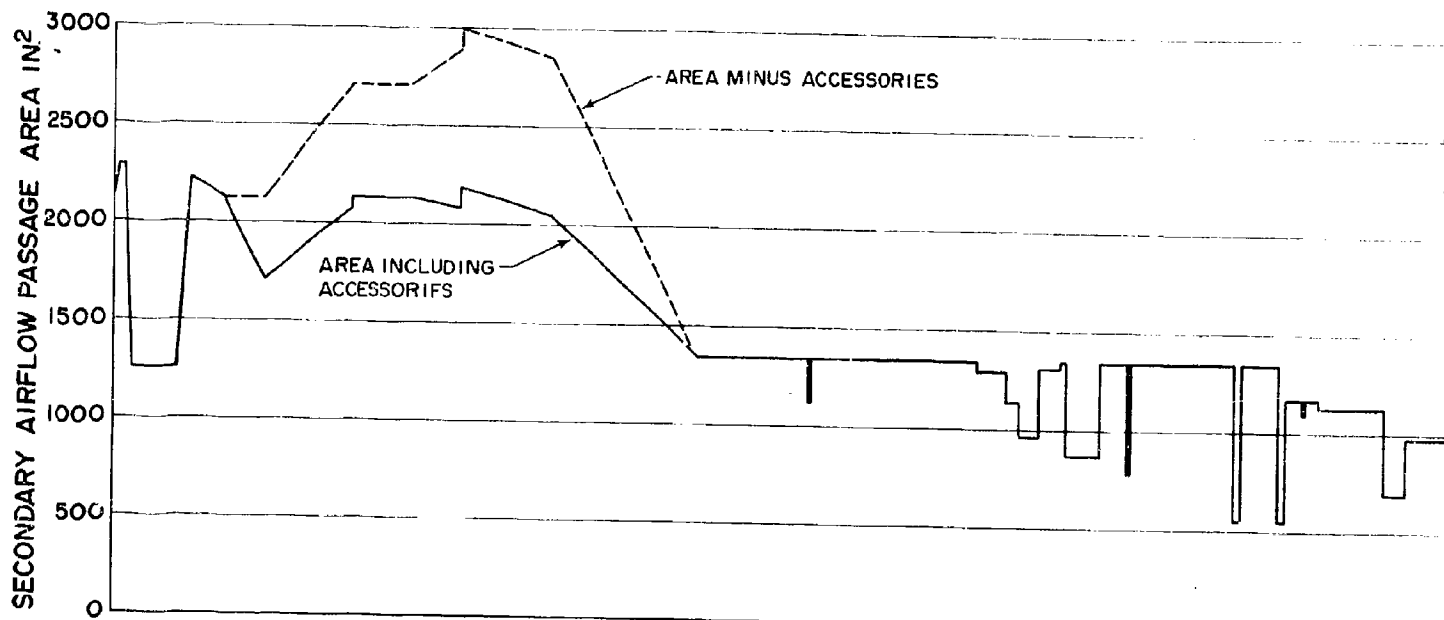


ARRANGEMENT OF FLAME ARRESTORS  
IN SECONDARY AIR PASSAGE

Figure 1-37

RECLASSIFIED AT 5 YEAR INTERVALS  
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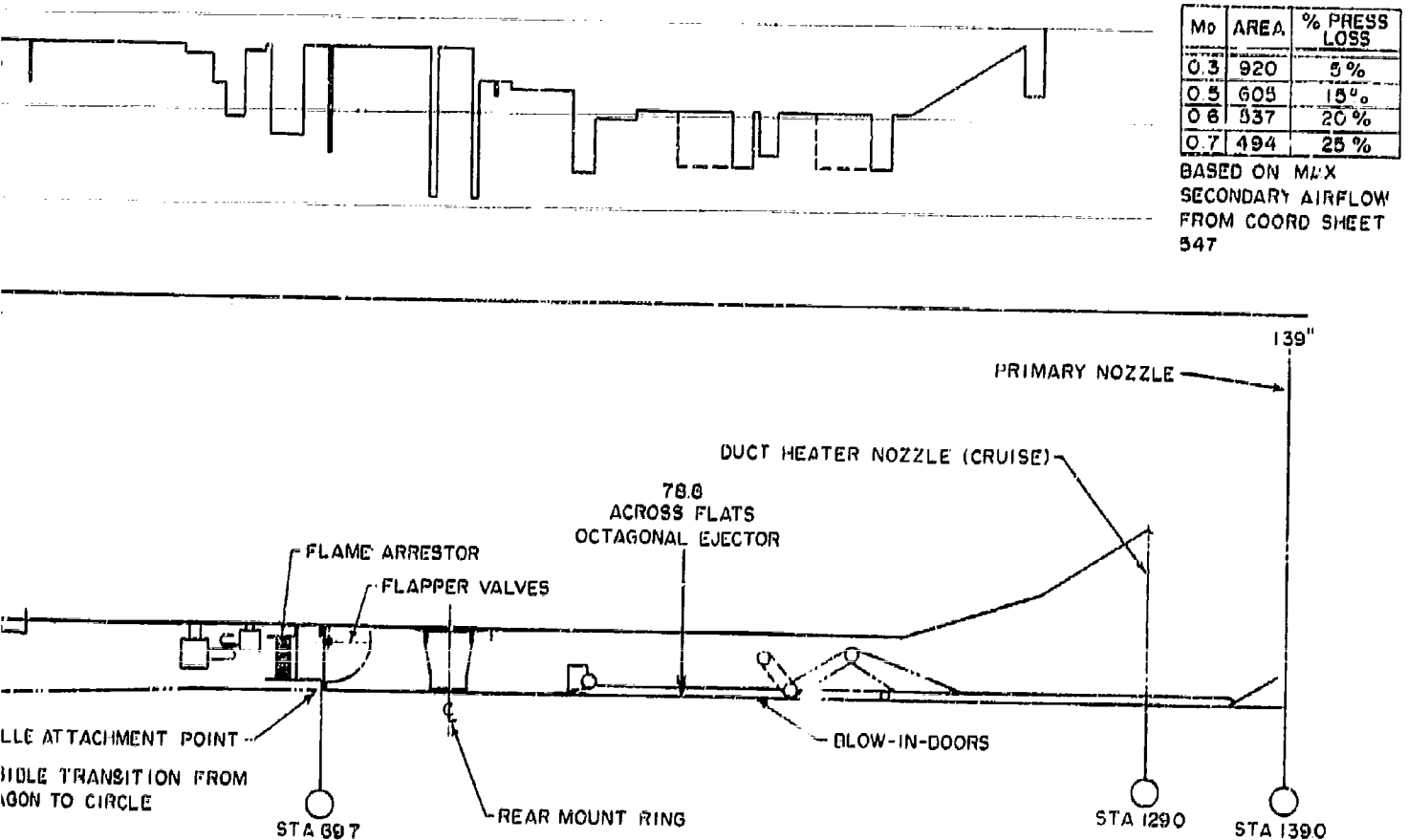


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ACCESSORIES



SECONDARY AIRFLOW PASSAGE GENERAL ARRANGEMENT

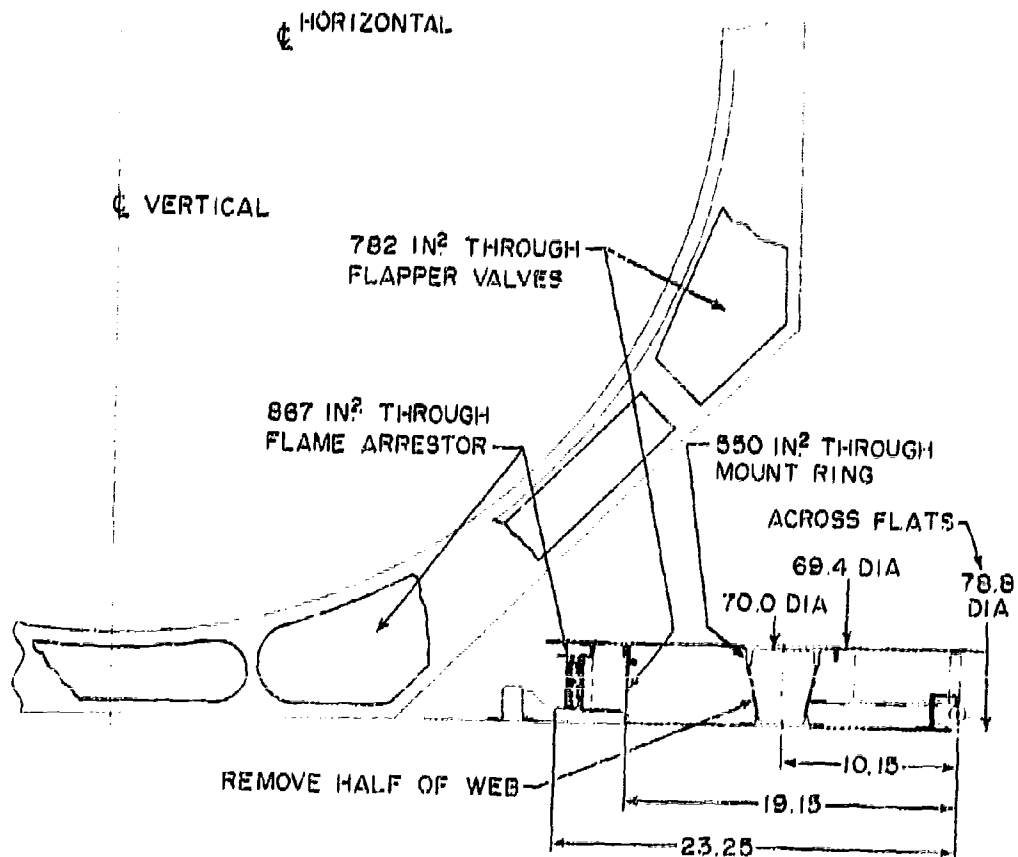
Figure 1-38

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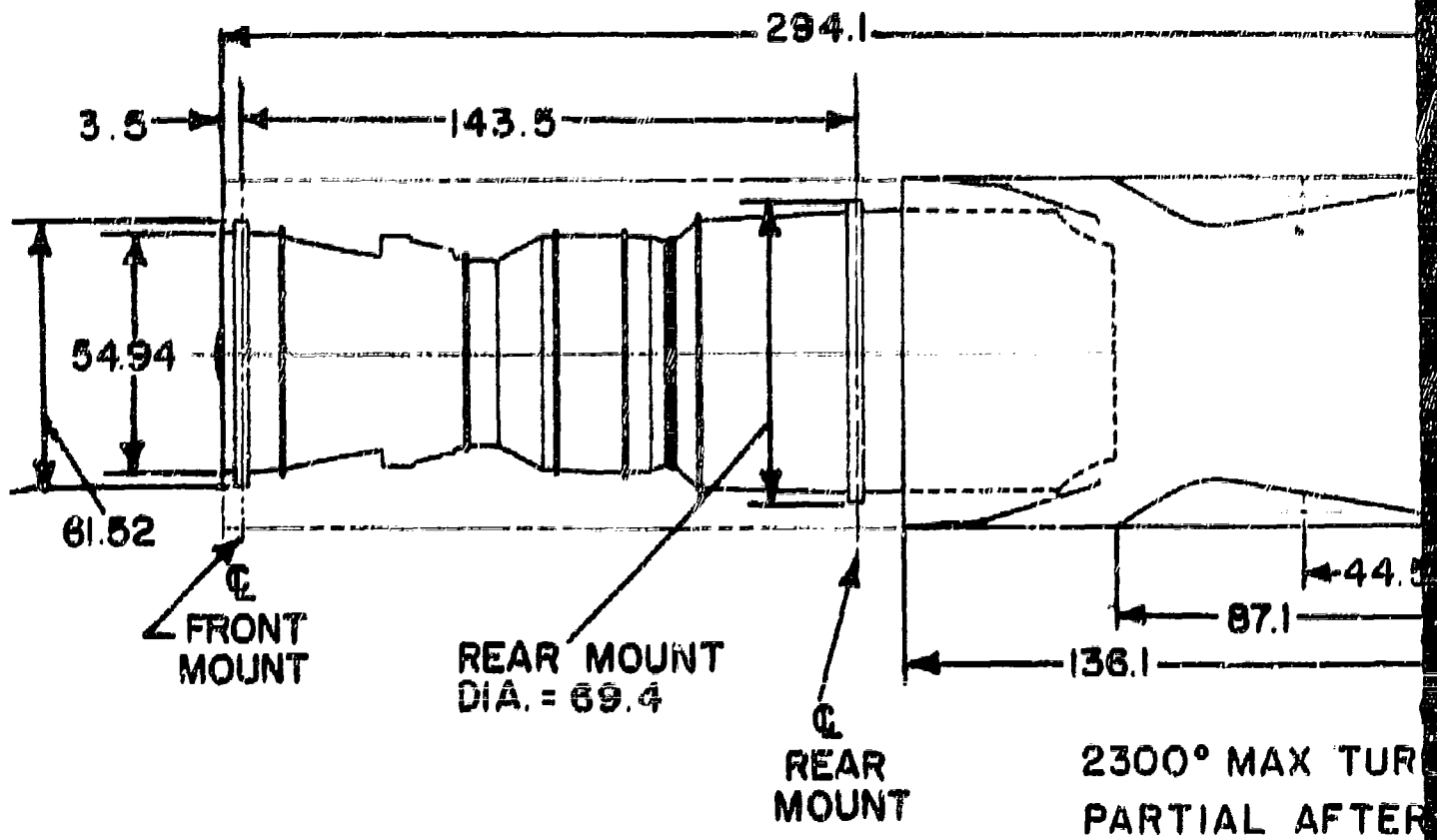
DETAILS OF SECONDARY AIRFLOW PASSAGE

Figure 1-39

Classified at 8 year interval  
Declassify after 12 years  
Date: 10/19/00

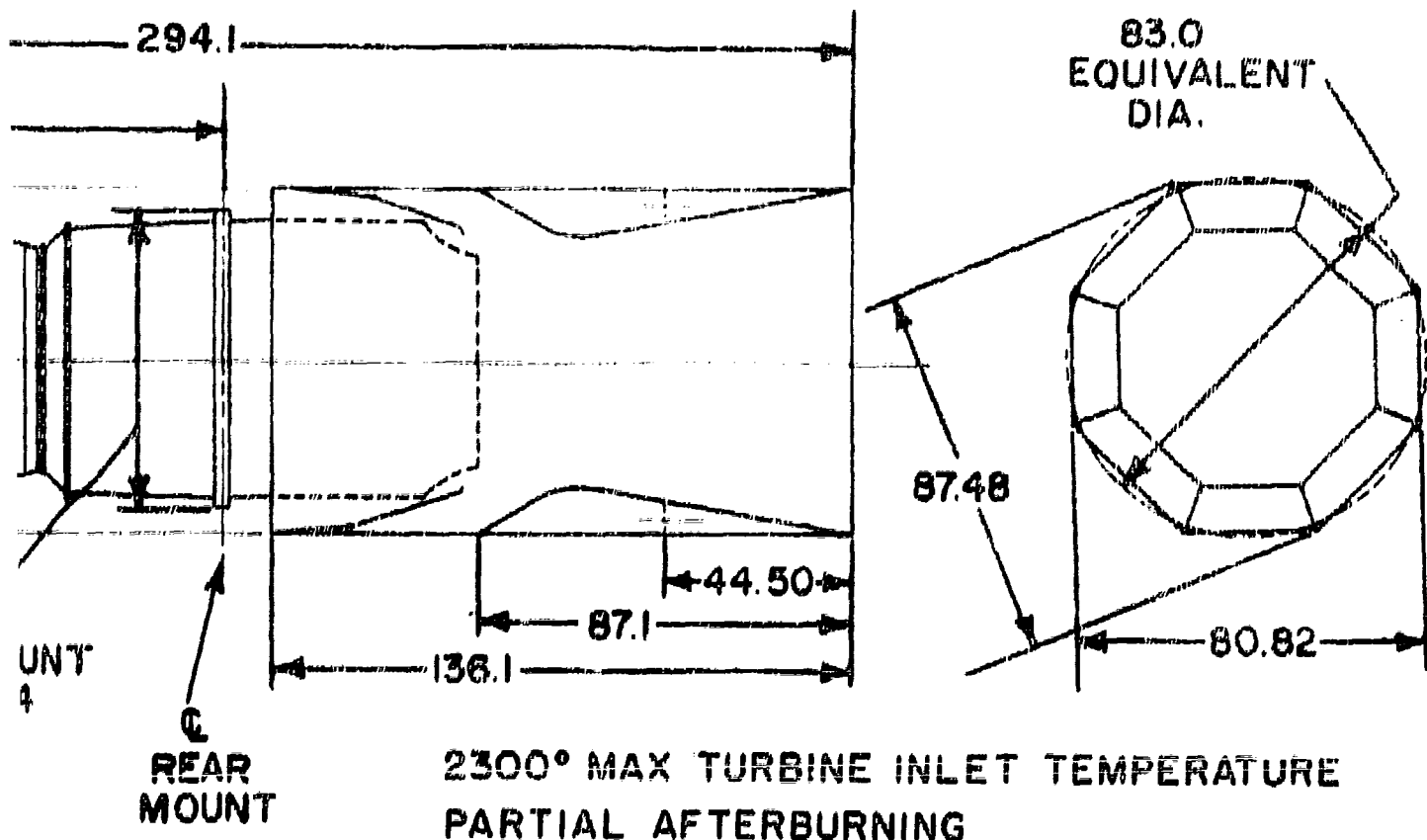
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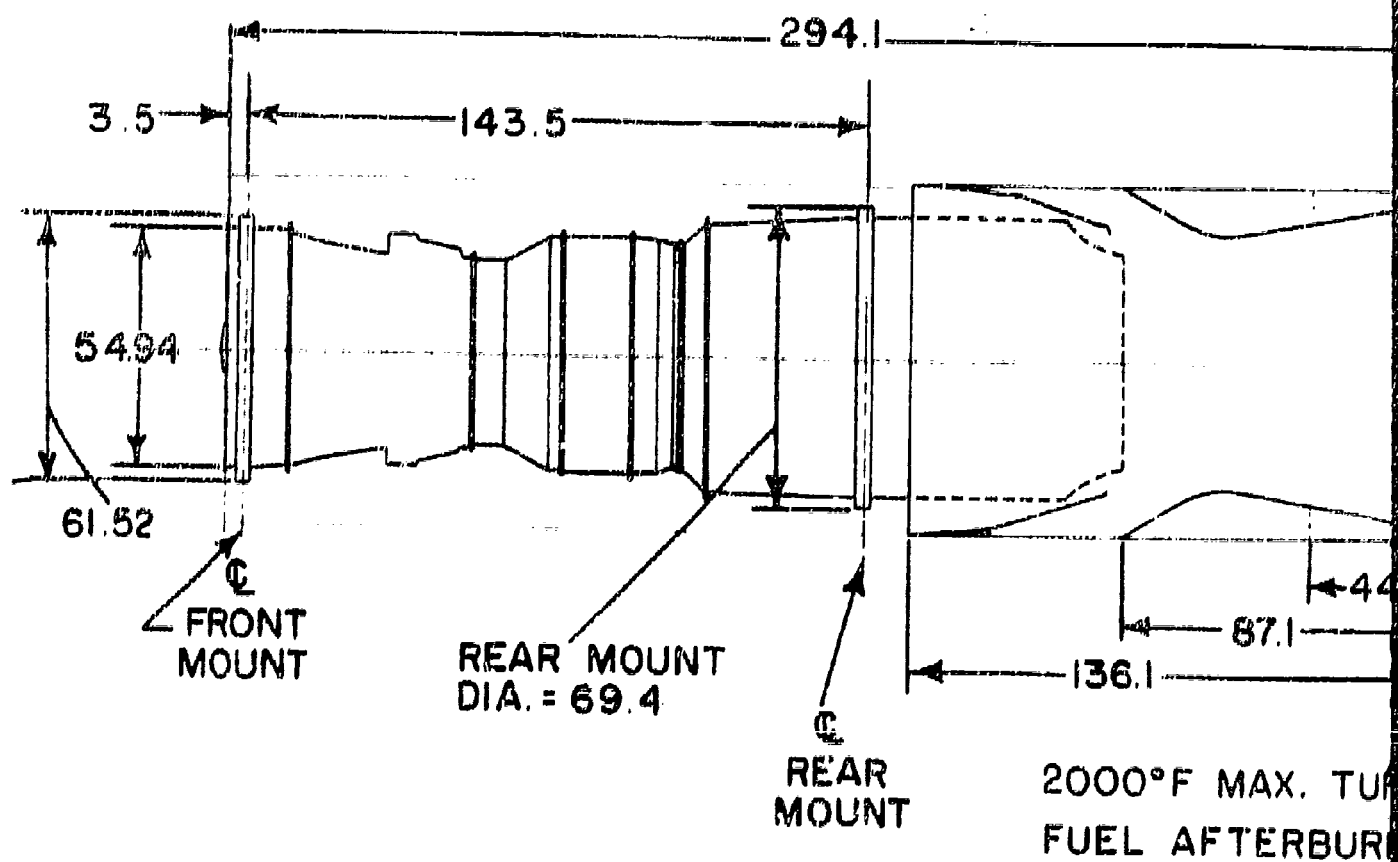
STJ227 525 LB

Fig.



BTJ227 525 LB/SEC TURBOJET

Figure 1-40



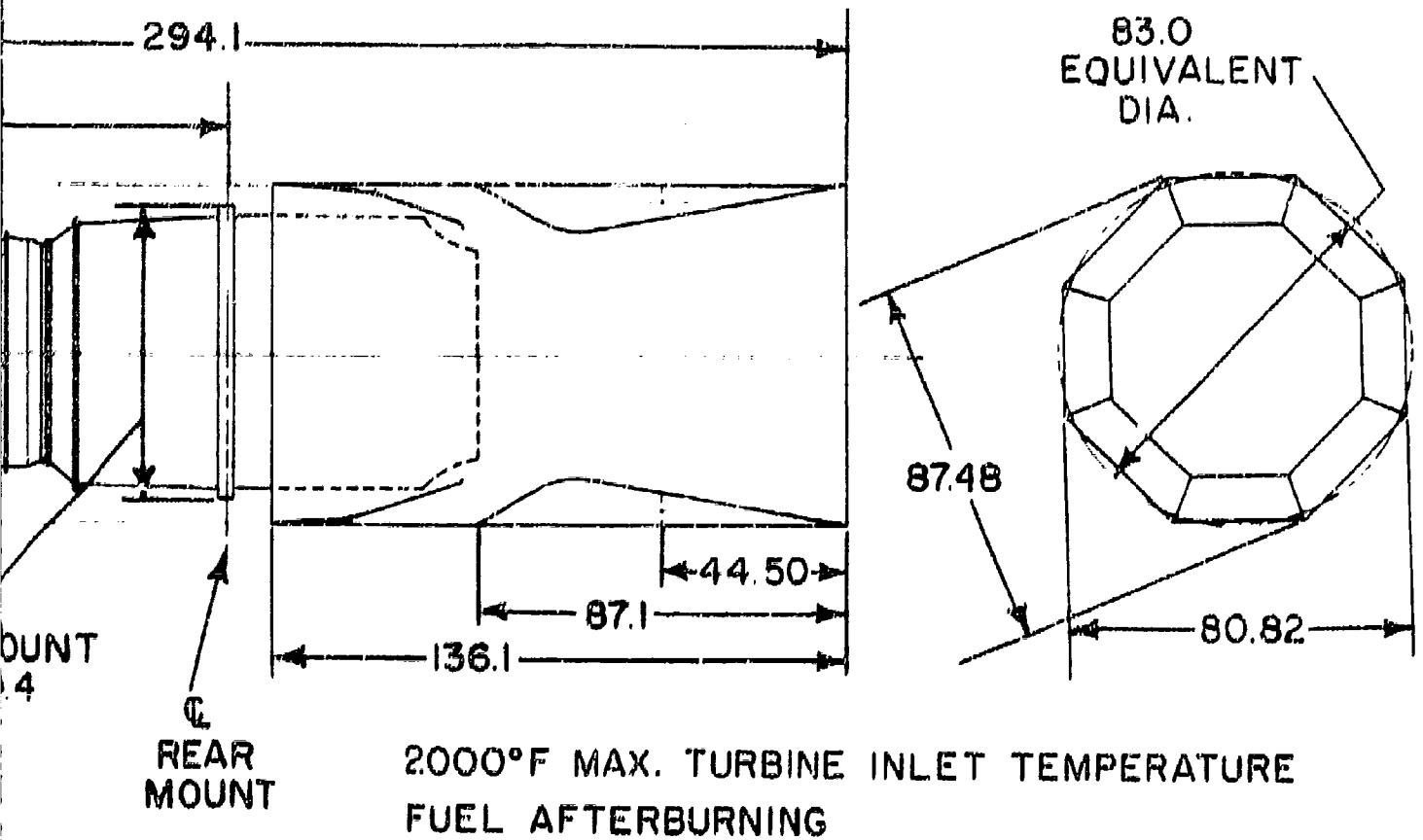
STJ227 525 1

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2

STJ227 525 LB/SEC TURBOJET

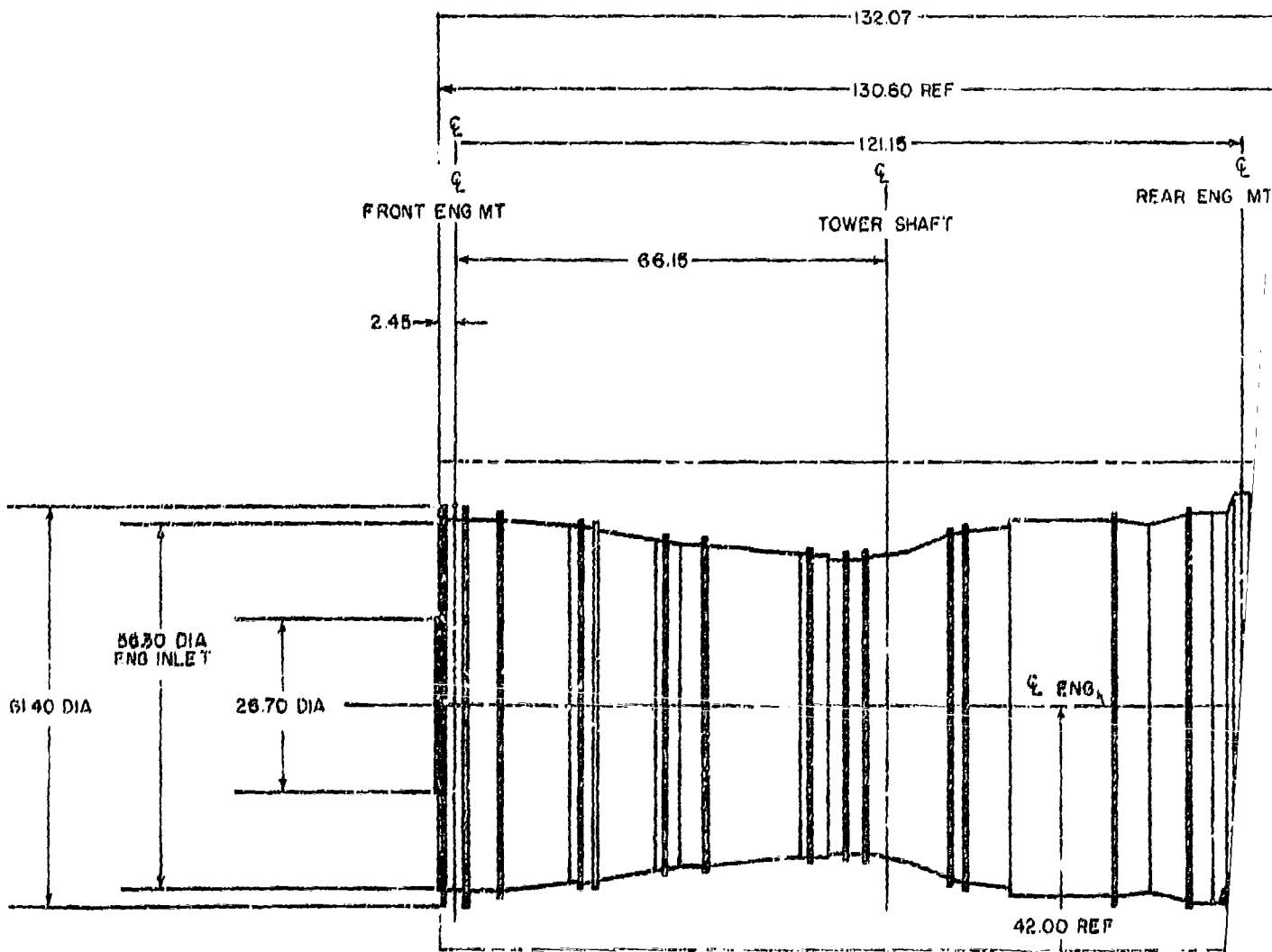
Figure 1-41

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EXAMINED BY 8 TECH DIVISION  
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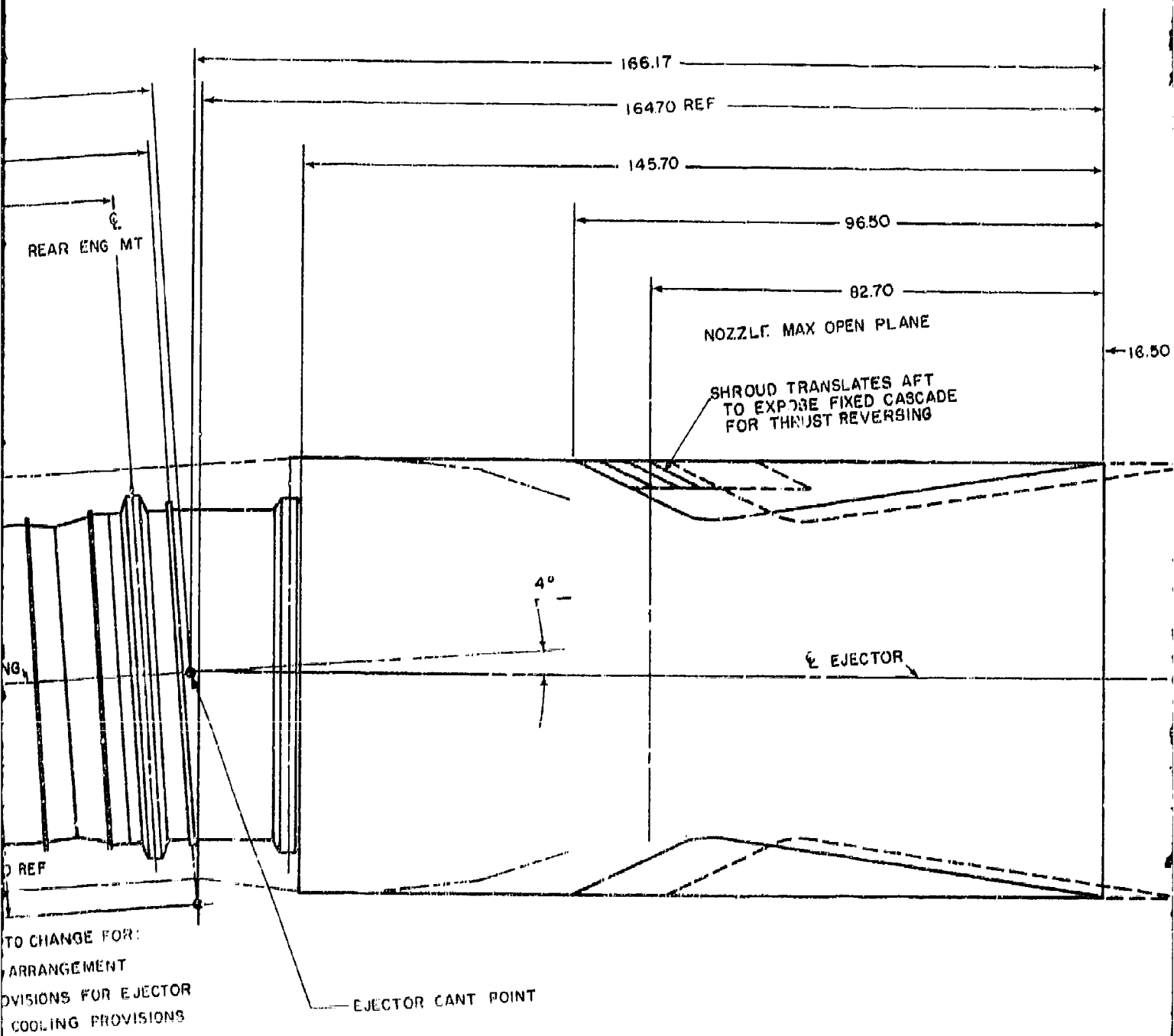
PRATT & WHITNEY AIRCRAFT



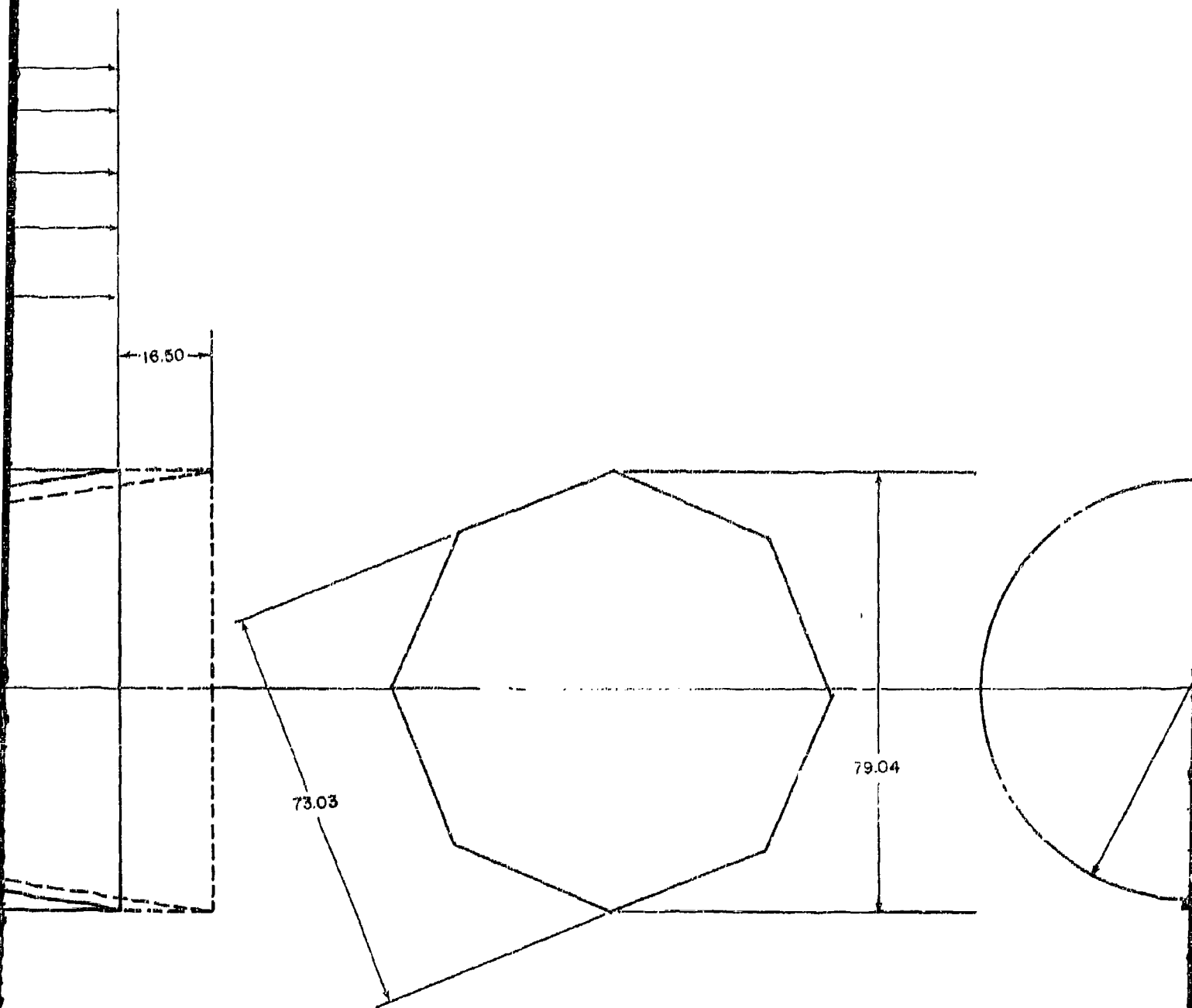
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1. ENGINE ACCESSORY ARRANGEMENT
2. SECONDARY AIR PROVISIONS FOR ENGINE
3. REAR CASE & A/B COOLING PROVISIONS

1



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TYPICAL TURBOJET ENGINE. FULL AFTERBU

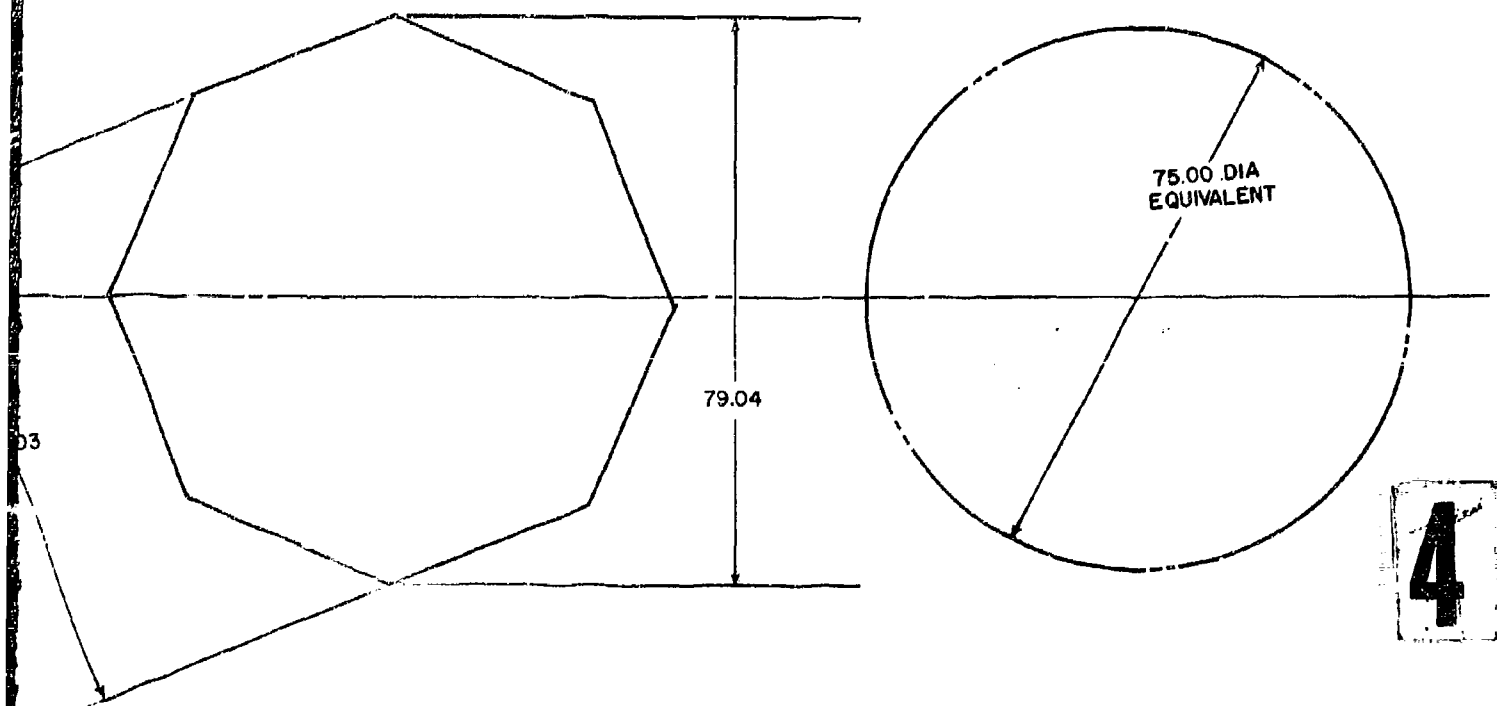
Figure 1-42

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TYPICAL TURBOJET ENGINE. FULL AFTERBURNING.

Figure 1-42

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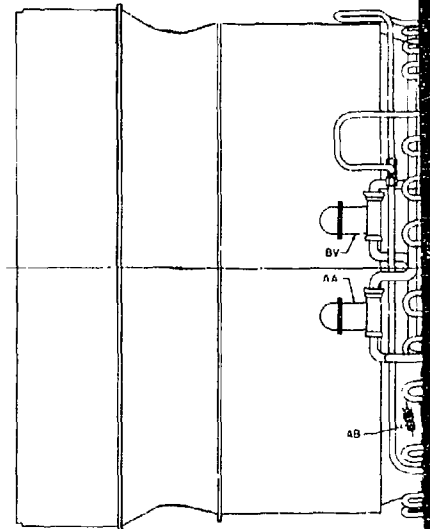
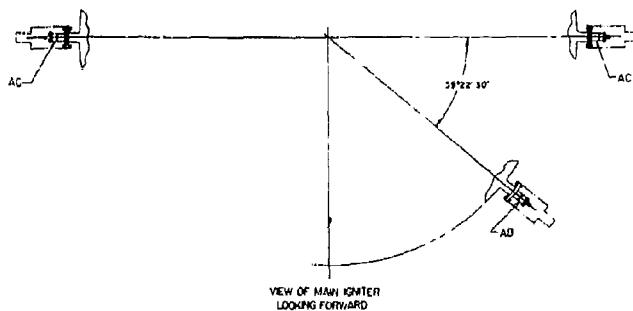
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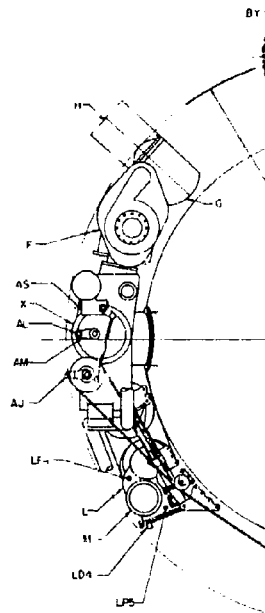
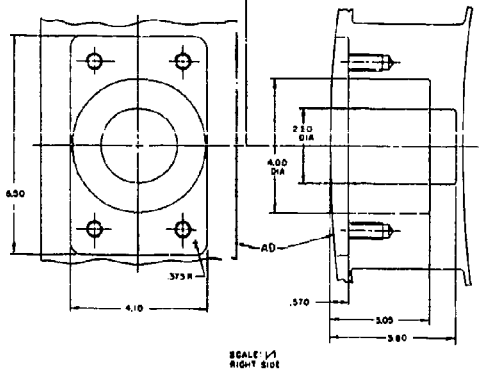
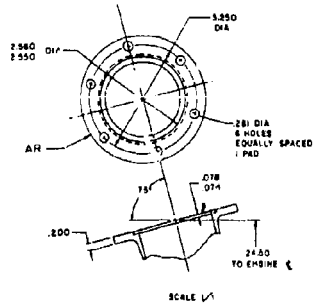
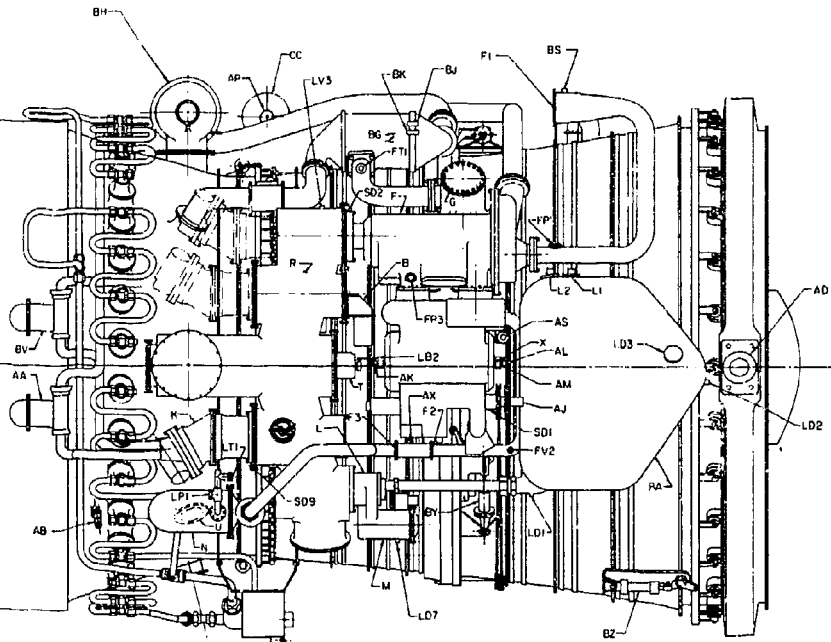
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# PRATT & WHITNEY AIRCRAFT

ZONE		ACCESSORY DRIVE PADS	
257, 277, 306	A	POWER TAKEOFF	TACHOMETER MOUNTING AND INFLUENCE BOSS (S)
FUEL DRAIN			
17C	FD1	COMBUSTION CHAMBER FUEL DRAIN	
18C	FD2	DUMP VALVE DRAIN (MAIN)	
19C	FD3	DUMP VALVE DRAIN (AFTERBURNER ZONE 1)	
20C	FD4	DUMP VALVE DRAIN (AFTERBURNER ZONE 2)	
21C	FD5	AFTERBURNER COMBUSTION CHAMBER FUEL DRAIN	
FUEL PRESSURE			
29A	FP1	FUEL PUMP INLET PRESSURE	
30E	FP3	HECK VALVE FUEL PRESSURE (OUTLET PRESSURE)	
FUEL FLOW			
197, 30F	F1	FUEL PUMP & AFTERBURNER PUMP SUPPLY INLET	
30D	F2	MAIN FUEL FLOWMETER SUPPLY INLET	
30D	F3	MAIN FUEL FLOWMETER SUPPLY OUTLET	
19E	F4	AFTERBURNER FLOWMETER SUPPLY INLET	
19D	F5	AFTERBURNER FLOWMETER SUPPLY OUTLET	
FUEL VENT			
29D	FV2	FUEL PUMP OUTLET VENT	
FUEL TEMPERATURE			
30E	FT1	HEATER OUTLET FUEL TEMPERATURE	
OIL BREATHER			
30D	LB2	MAIN OIL OVERBOARD BREATHER	
OIL DRAIN			
29C	LD1	OIL TANK DRAIN	
28D	LD2	OIL TANK OVERFLOW DRAIN	
29D	LD3	OIL CUP OVERFLOW DRAIN	
29E	LD4	GEARBOX MAIN OIL DRAIN	
30C	LD7	OIL STRAINER DRAIN	
OIL FLOW			
29E	LI	OIL TANK NEARBY FILLER	
29E	LI2	OIL TANK MANUAL FILL	
OIL PRESSURE			
29D	LP1	PRESSURE FOR TRANSDUCER	
29E	LP4	OIL FILTER INLET PRESSURE—PROVISIONS FOR PRESSURE	
29C	LP5	OIL FILTER OUTLET PRESSURE—DIFFERENTIAL	
OIL TEMPERATURE			
30D	LT1	MAIN OIL TEMPERATURE	
OIL VENT			
30F	LV3	OIL PRESSURE TRANSDUCER VENT	
SEAL DRAIN			
29D	SD1	FUEL CONTROL SEAL DRAIN	
30E	SD2	FUEL PUMP SEAL DRAIN	
30D	SD3	HYDRAULIC PUMP SEAL DRAIN	
18D	SD14	AFTERBURNER FUEL PUMP SEAL DRAIN	
TEMPERATURE SENSING			
20D	TT5	EXHAUST EXIT TEMPERATURE (IN)	
20D	TTT5	TURBINE EXIT TEMPERATURE (INDIVIDUAL)	
PRESSURE SENSING			
18F	PTS	TURBINE EXIT PRESSURE	
MISCELLANEOUS			
24E, 30E	F	MAIN FUEL PUMP	
25E, 30E	G	FUEL PUMP FUEL FILTER DRAIN	
24E	H	FUEL PUMP FILTER (MIN SPACE FOR REMOVAL)	
19E, 23E	J	AFTERBURNER FUEL PUMP	
30D	K	HYDRAULIC PUMP	
24E, 30D	L	OIL PUMP	
24E, 30C	M	OIL FILTER	
30C	N	FUEL OIL COOLER (MAIN)	
19C, 25C	P	FUEL OIL COOLER (AFTERBURNER)	
30E	Q	GEARBOX	
30D	R	AUTOMATIC RESTART SWITCH	
30D	T	BREATHING PRESSURIZING VALVE	
18E, 19E, 21C	U	HIGH PRESSURE BLEED PAD	
24D, 29D	X	MAIN FUEL CONTROL	
18D, 21D	Y	AFTERBURNER FUEL CONTROL	
17D	Z	EXHAUST NOZZLE CONTROL	
32D	AA	HYDRAULIC DISCHARGE FILTER	
30D, 34B, 17C, 20D	AB	IGNITER PLUG (MAIN)	
30D, 31B, 18B	AC	IGNITER PLUG (AFTERBURNER)	
7-8, 24E, 20A, 20E	AD	ENGINE FRONT MOUNTING PROVISIONS	
19E, 18E, 18E	AE	ENGINE REAR MOUNTING PROVISIONS	
20F	AF	FUEL RETURN TO SUPPLEMENTARY CLOSING	

ZONE		GROUND HANDLING HOLES (FRONT MOUNT)	
21A	AG	GROUND HANDLING AREA (REAR)	
11C	AH	POWER CONTROL LEVER (90° ANGLE OF TRAVEL)	
28D	AJ	SHUT OFF LEVER (90° ANGLE OF TRAVEL)	
30D	AK	APPROACH VELOCITY CONTROL LEVER (90° ANGLE OF TRAVEL)	
28D	AL	WACH NO. ON SHOCK POSITION RESET LEVER (90° ANGLE OF TRAVEL)	
28D, 29D	AM	AIR TURBINE VALVE (AFTERBURNER FUEL PUMP)	
19E	AN	IGNITION EXCITER ELECTRICAL CONN.	
31F	AP	THERMAL ANTI-ICING PAD	
31B	AR	AEROHYDRAULIC BRAKE CONTROL AIR SUPPLY CONN.	
24E, 28D	AS	EXHAUST NOZZLE FEEDBACK	
17D	AT	CHECK & DUMP VALVE (AFTERBURNER ZONE 1)	
18C	AU	CHECK & DUMP VALVE (AFTERBURNER ZONE 2)	
18C	AV	MAIN FUEL FLOWMETER MOUNTING PROVISIONS	
19D	AW	AFTERBURNER FUEL FLOWMETER MOUNTING PROVISIONS	
30D	AX	FUEL CONTROL FUEL FILTER (MIN SPACE FOR REMOVAL)	
21E, 18E	AY	WINDMILL BYPASS CHECK & DUMP LOW OIL VALVE (MAIN)	
18C, 25C	AZ	OIL TANK	
29D	BA	NOZZLE POSITION INDICATOR MOUNTING PROVISIONS	
17D	BB	REVERSE POSITION INDICATOR CONN. (ELECTRICAL)	
20D	BC	THERMOSTAT FUEL BYPASS VALVE	
19C	BD	VIBRATION MOUNT (SPACE RESERVED FOR PICKUP PROVIDED BY AIRFRAME MFG)	
20E	BE	FUEL HEATER	
17E, 24E, 31F	BH	POWER TAKEOFF DEARBON	
30F	BI	FUEL HEATER VALVE ELECTRICAL CONN.	
30F	BJ	FUEL HEATER VALVE POSITION INDICATOR (ELECTRICAL)	
19F	BL	POWER TAKEOFF DECOUPLER	
18F, 24F	BN	POWER TAKEOFF DECOUPLER ACTUATING SHAFT	
24D	BO	CONTROL CABLE—FUEL CONTROLS	
18F, 28F	BS	FUEL INLET TEMPERATURE SENSOR	
18D	BU	AFTERBURNER FUEL OIL COOLER THERMOSTATIC BYPASS VALVE	
32D	BV	HYDRAULIC RETURN FILTERS	
18E	BW	EXHAUST NOZZLE ACTUATORS	
21D, 28D, 29F, 25F, 30C	BY	STARTING BLEED DOOR ACTUATORS	
20E, 29C	BZ	INLET GUIDE VANE ACTUATORS	
18C, 18F	CA	AEROHYDRAULIC BRAKE ACTUATORS	
20D	CB	EXCITER (MAIN)	
18F, 30F	CC	EXCITER (AFTERBURNER)	

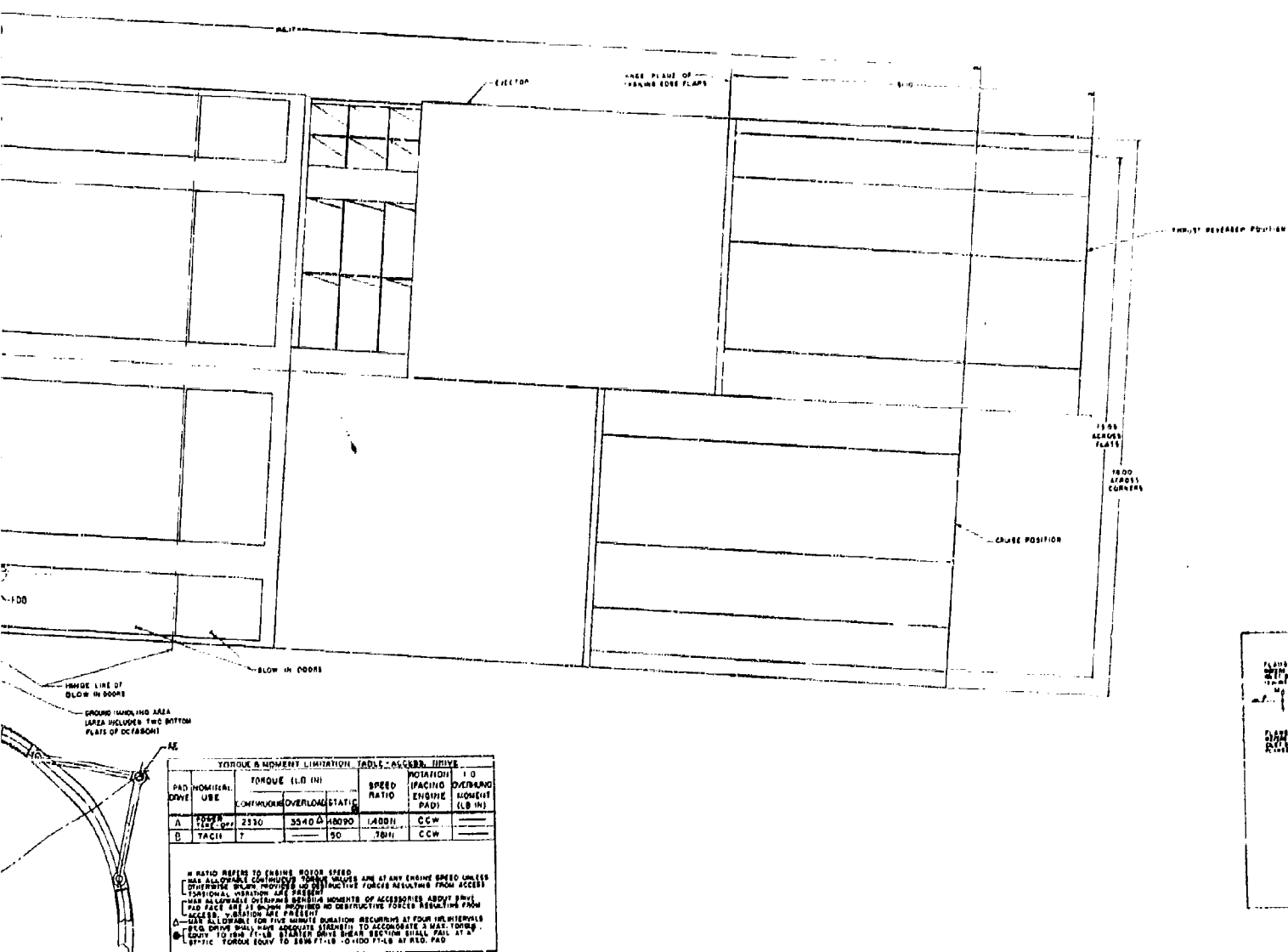








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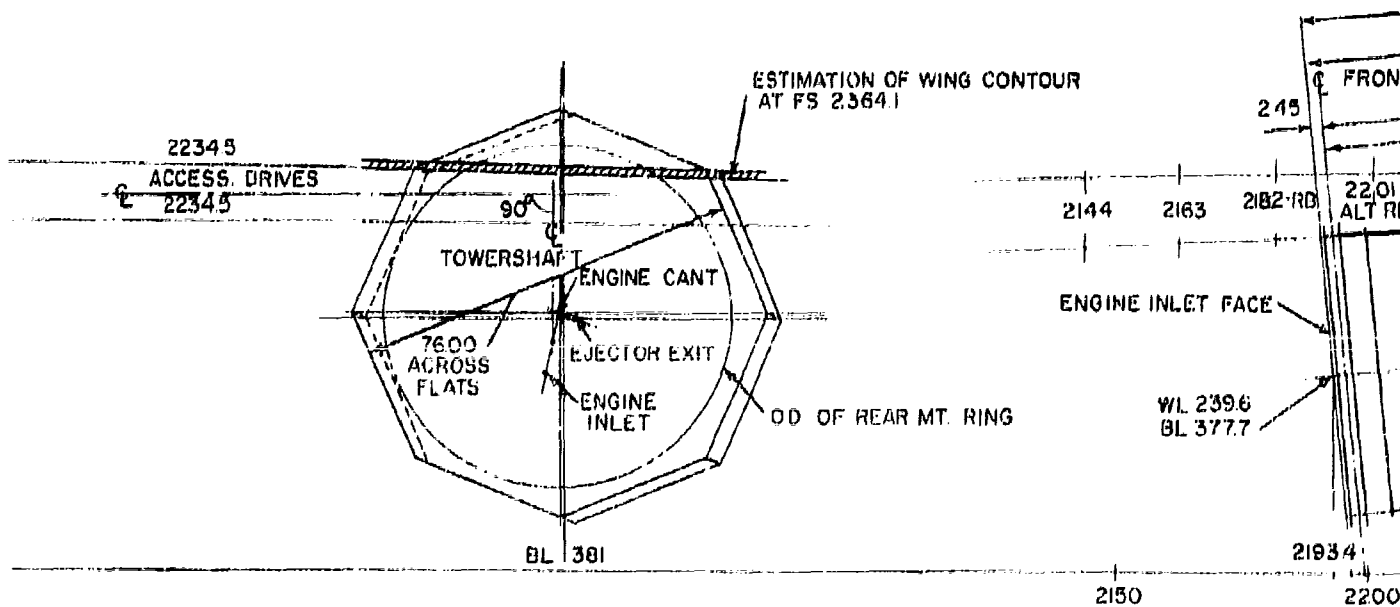
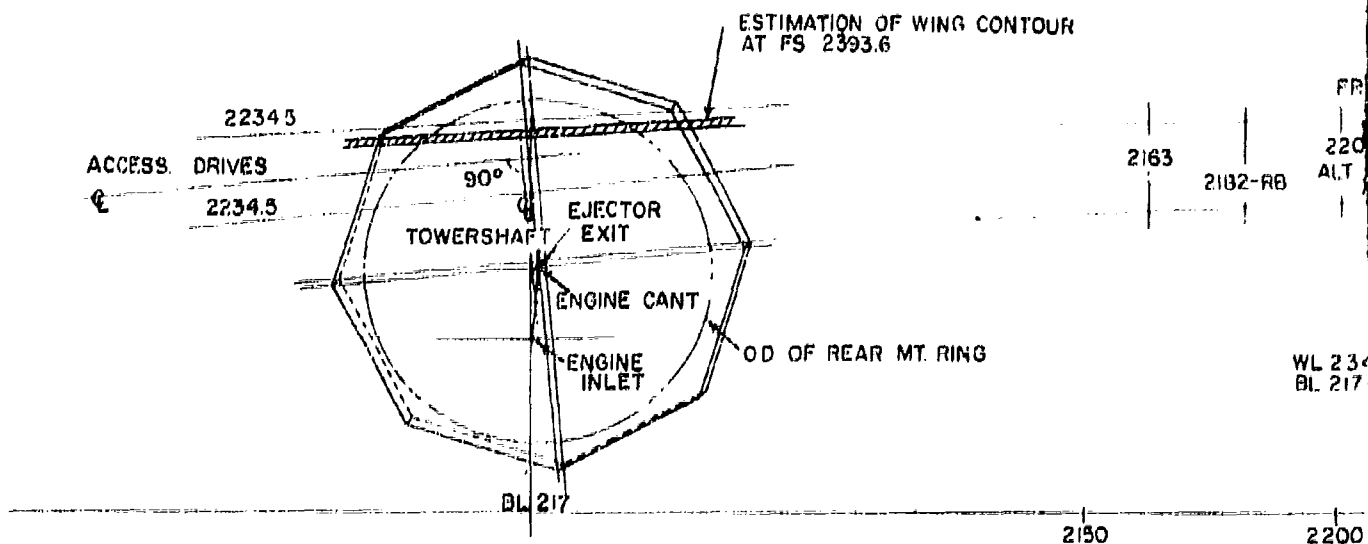
PROPOSED TURBOJET ACCESSORY ARR

Figure 1-43

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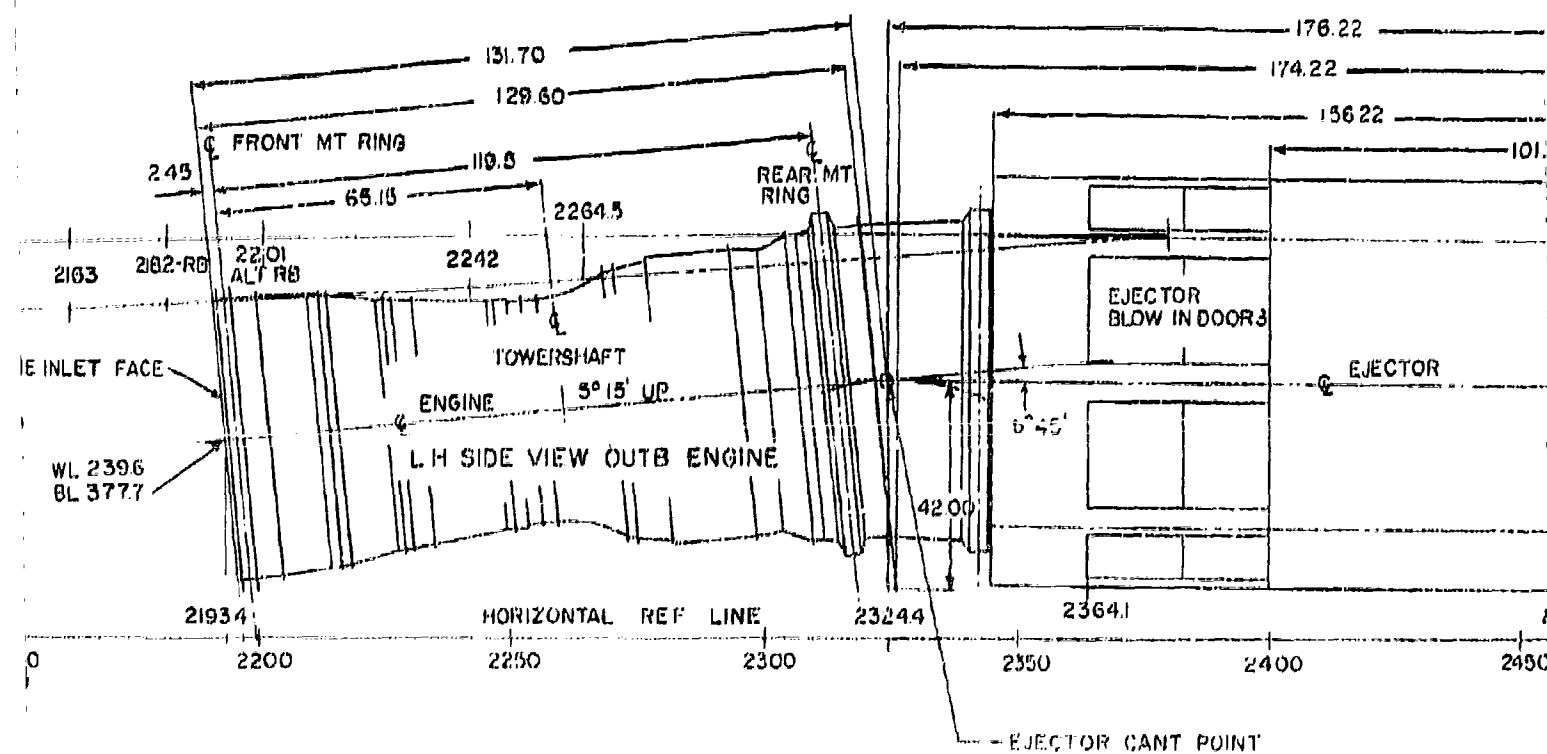
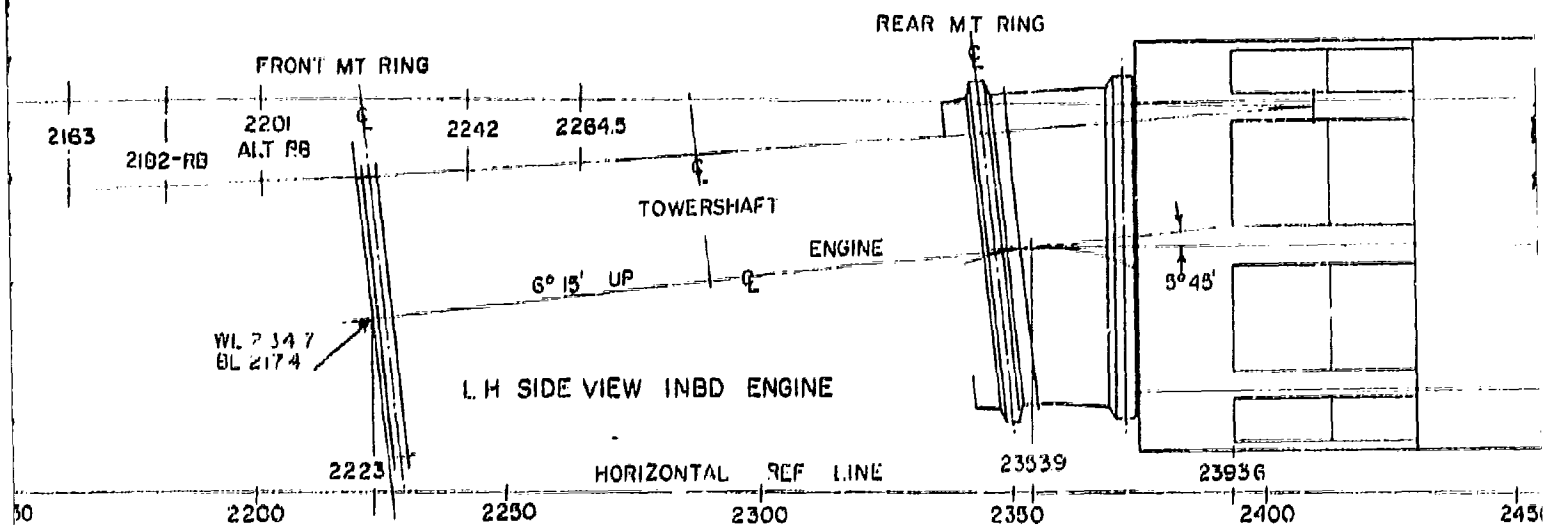
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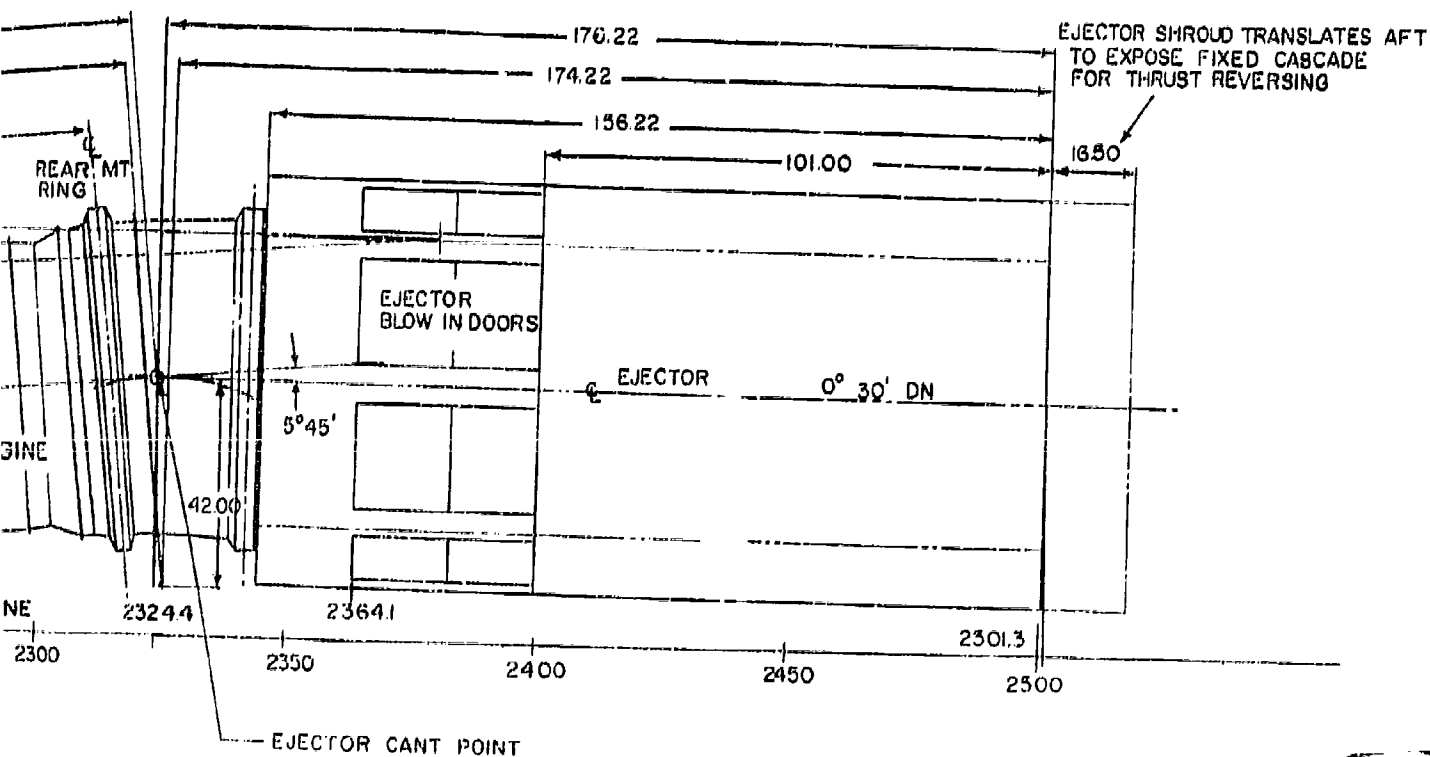
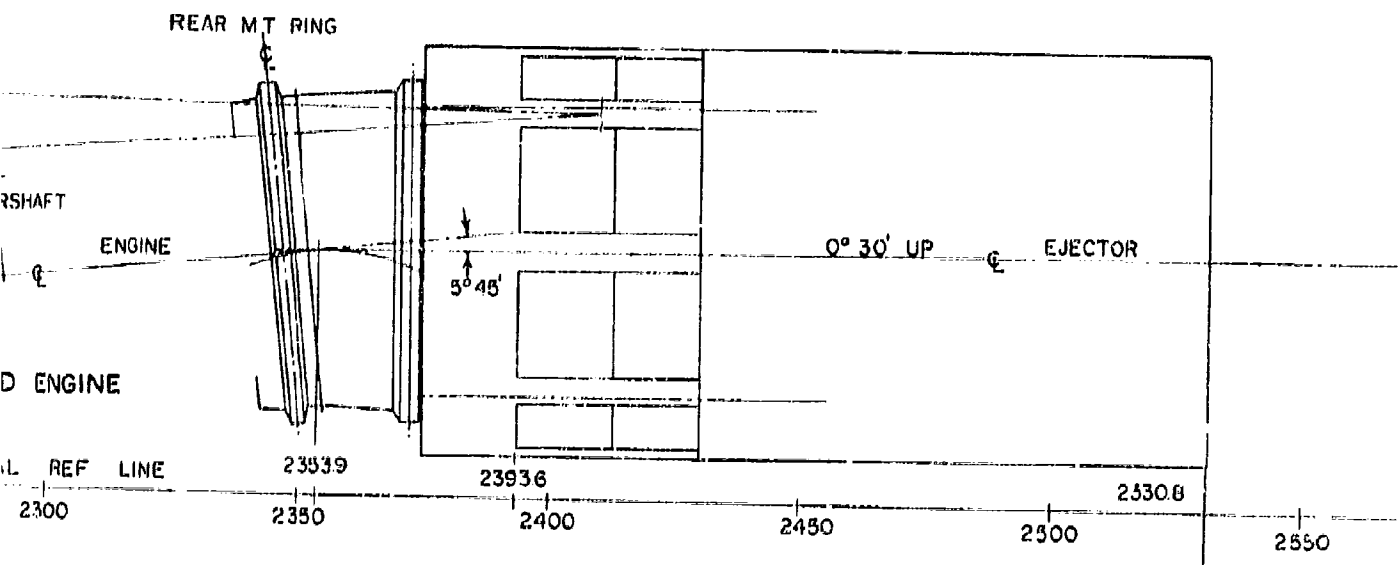
TYPICAL TURBOJET INST  
EFFECT OF INCREASE

Figure 1

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TYPICAL TURBOJET INSTALLATION SHOWING  
EFFECT OF INCREASED CANT ANGLE

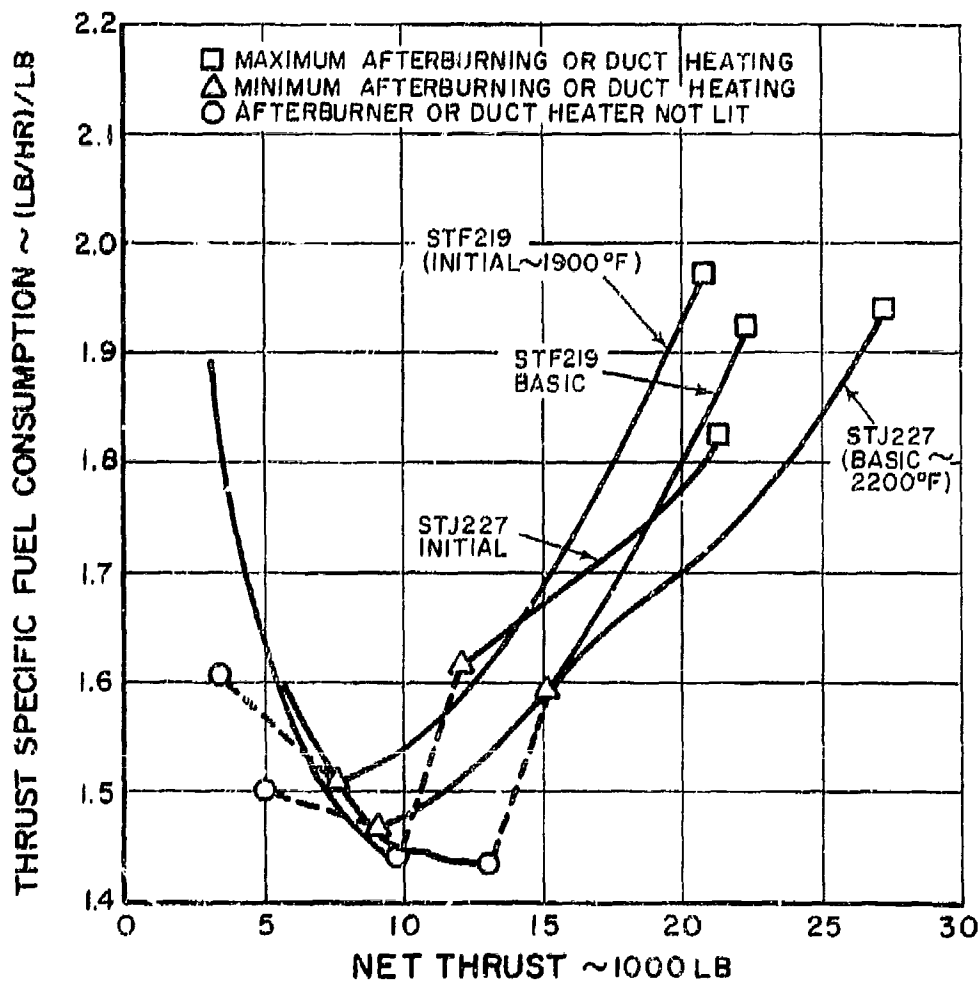
Figure 1-44

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US STANDARD ATMOSPHERE - 1962 (GEOM)  
 RAM RECOVERY PER MIL-E-5008B  
 65,000 FT PHASE II B MN 2.7  
 BASE AIRFLOW



ESTIMATED PERFORMANCE OF STF219 AND STJ227 ENGINES  
 AT MACH 2.7 AT 65,000 FEET

Figure 1-45

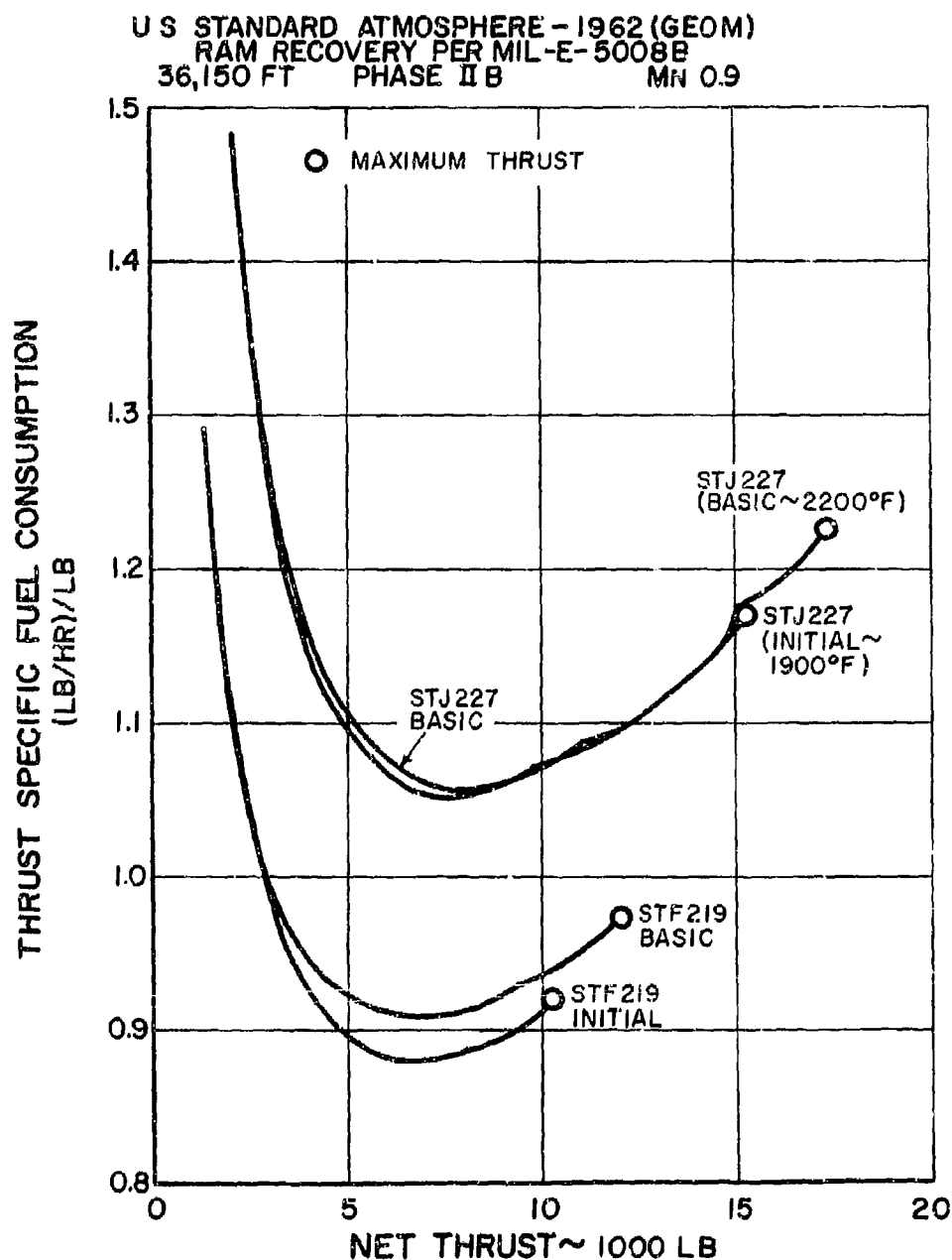
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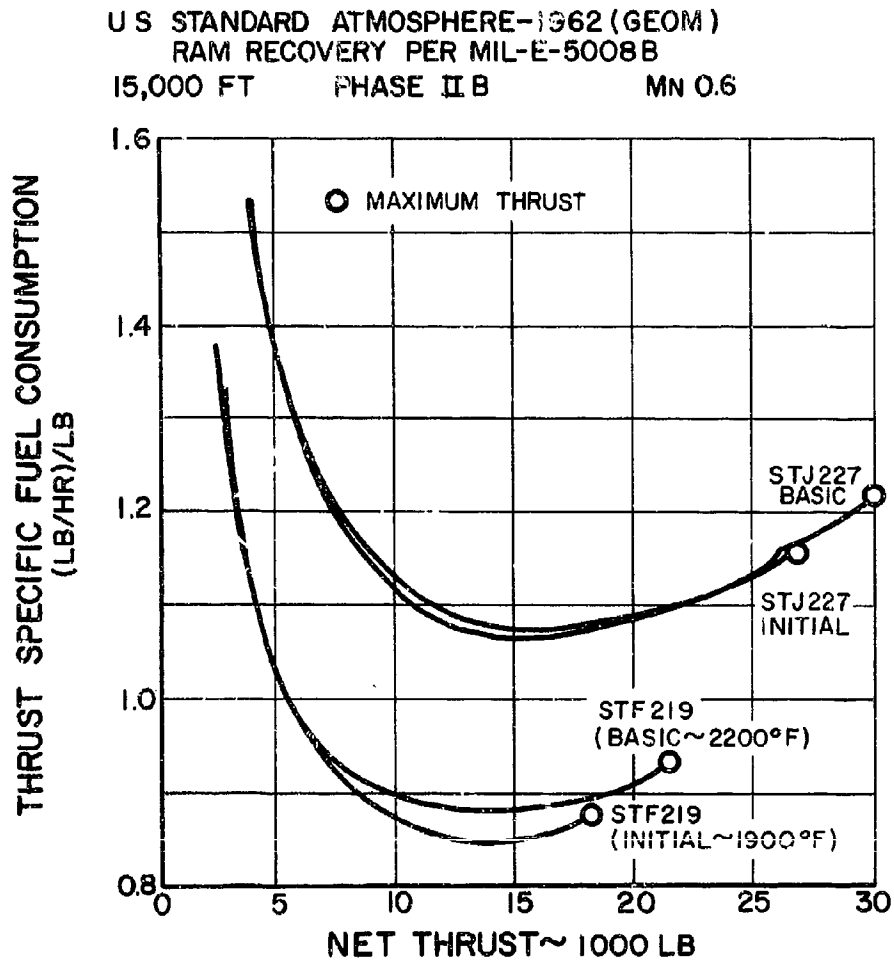
ESTIMATED PERFORMANCE OF STF219 AND STJ227 ENGINES  
 AT MACH 0.9 AT 36,150 FEET

Figure 1-46

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ESTIMATED PERFORMANCE OF STF219 AND STJ227 ENGINES  
AT MACH 0.6 AT 15,000 FEET

Figure 1-47

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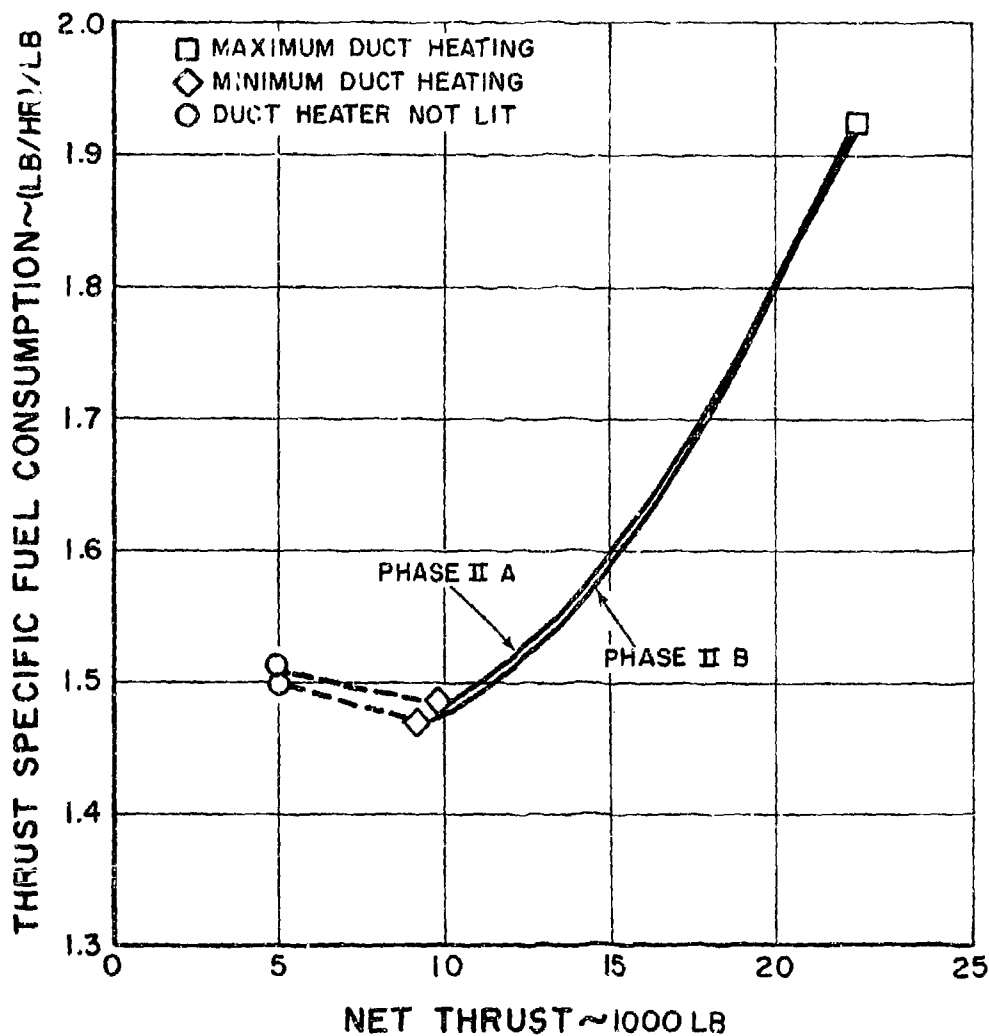
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US STANDARD ATMOSPHERE - 1962 (GEOM)  
RAM RECOVERY PER MIL-E-5008B  
DESIGN TURBINE INLET TEMPERATURE 2200°F  
65,000 FT Mn 2.7



ESTIMATED PERFORMANCE OF STF219 ENGINE AFTER PHASE IIA  
AND AFTER PHASE IIB AT MACH 2.7 AT 65,000 FEET

Figure 1-48

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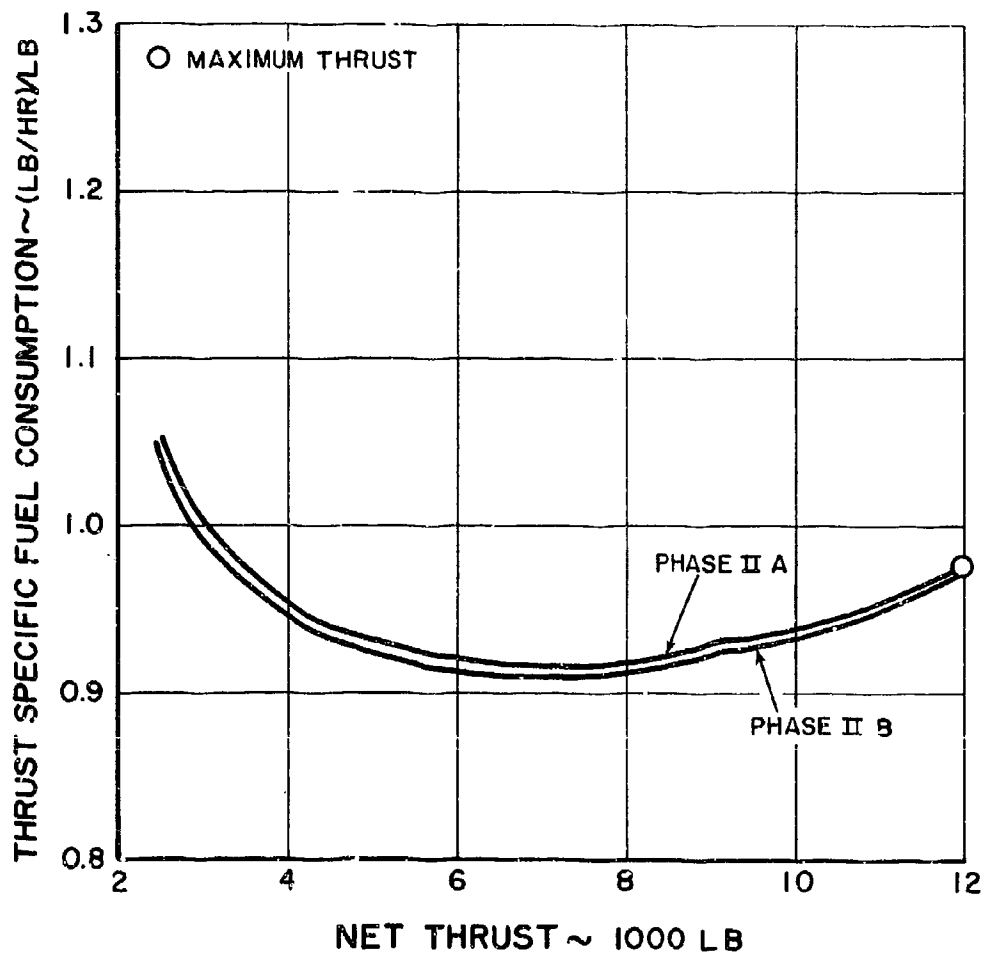
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U S STANDARD ATMOSPHERE - 1962 (GEOM)  
RAM RECOVERY PER MIL-E-5008 B  
DESIGN TURBINE INLET TEMPERATURE 2300°F  
36,150 FT Mn 0.9



ESTIMATED PERFORMANCE OF STF219 ENGINE AFTER PHASE IIA  
AND AFTER PHASE IIB AT MACH 0.9 AT 36,150 FEET

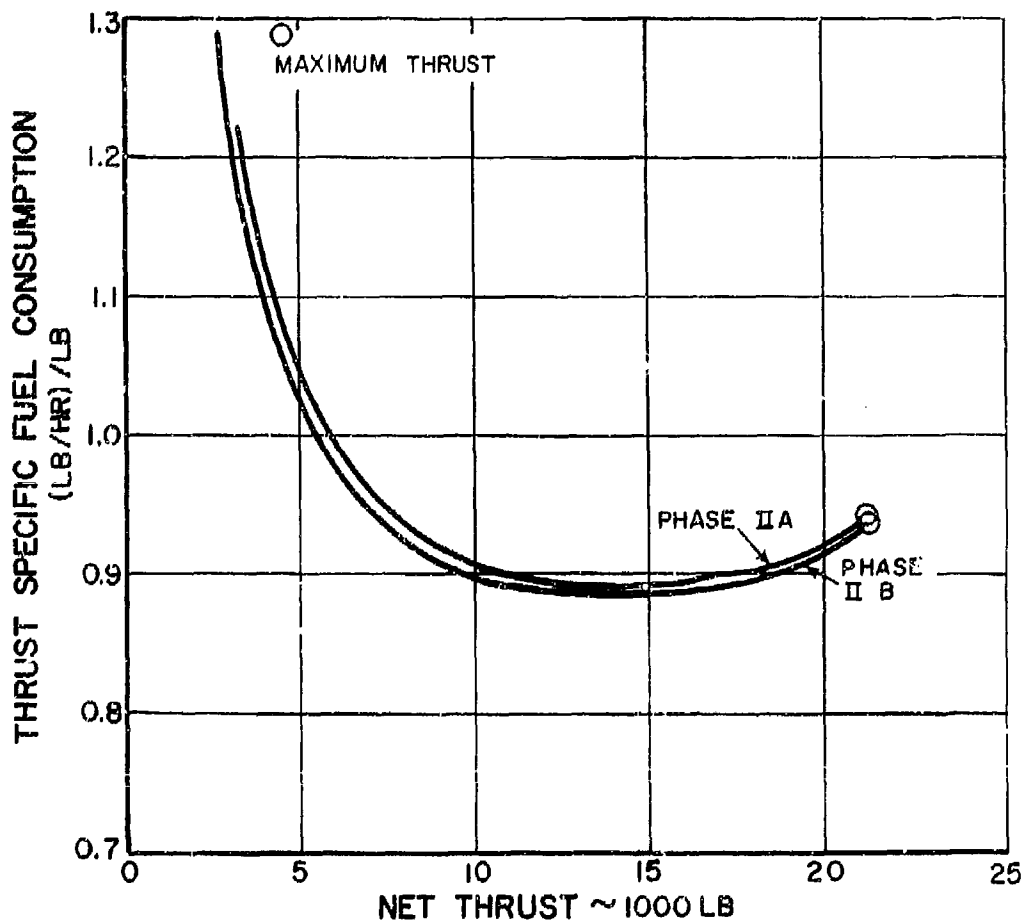
Figure 1-49

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US STANDARD ATMOSPHERE - 1962 (GEOM)  
RAM RECOVERY PER MIL-E-5008 B  
DESIGN TURBINE INLET TEMPERATURE - 2300 °F  
15,000 FT Mn 0.6



ESTIMATED PERFORMANCE OF STF219 ENGINE AFTER PHASE IIA  
AND AFTER PHASE IIB AT MACH 0.6 AT 15,000 FEET

Figure 1-50

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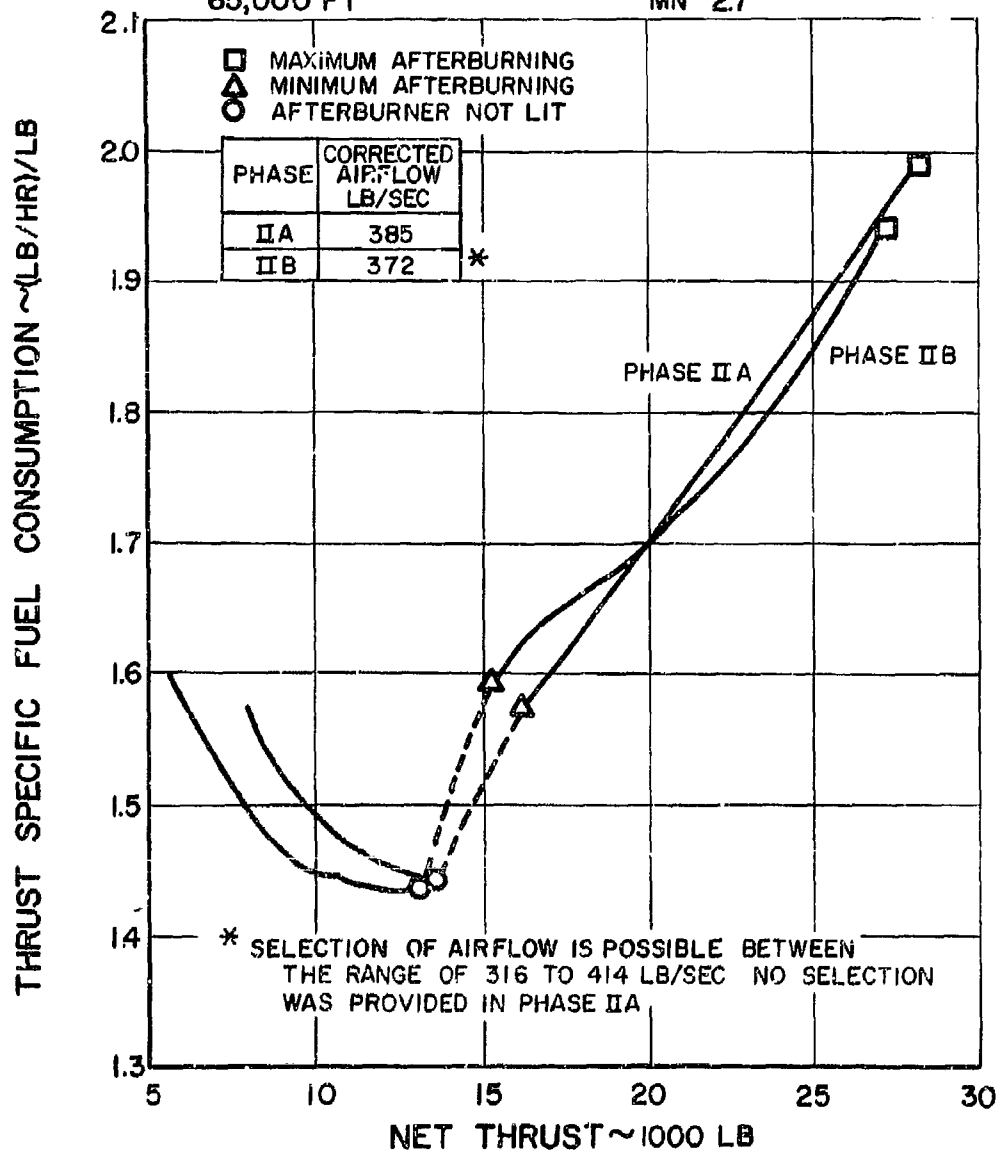
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US STANDARD ATMOSPHERE - 1962 (GEOM)  
 RAM RECOVERY PER MIL-E-5008B  
 DESIGN TURBINE INLET TEMPERATURE 2200°F  
 65,000 FT MN 2.7



ESTIMATED PERFORMANCE OF STJ227 ENGINE AFTER PHASE IIA  
 AND AFTER PHASE IIB AT MACH 2.7 AT 65,000 FEET

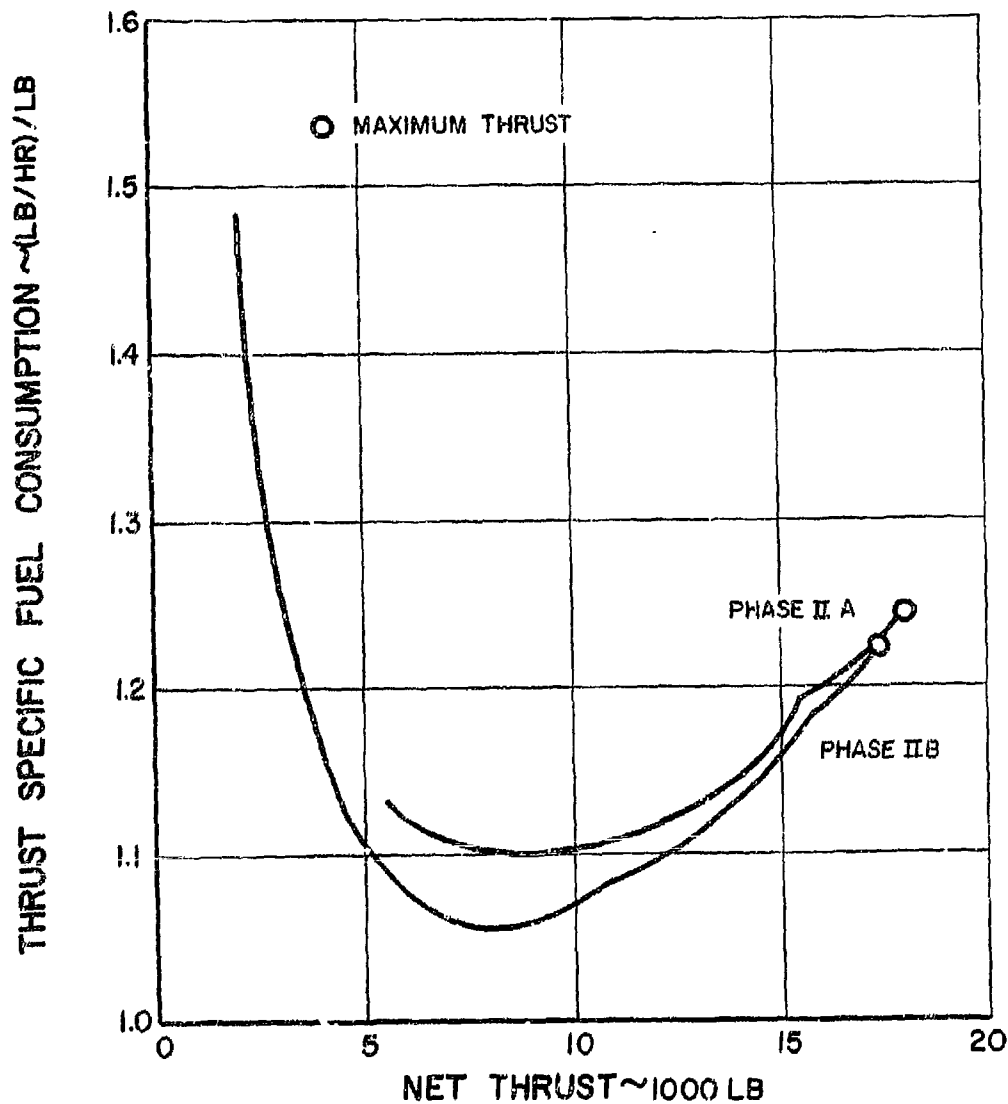
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U S STANDARD ATMOSPHERE - 1962 (GEOM)  
 RAM RECOVERY PER MIL-E-5008B  
 DESIGN TURBINE INLET TEMPERATURE 2300°F  
 36,150 FT Mn 0.9



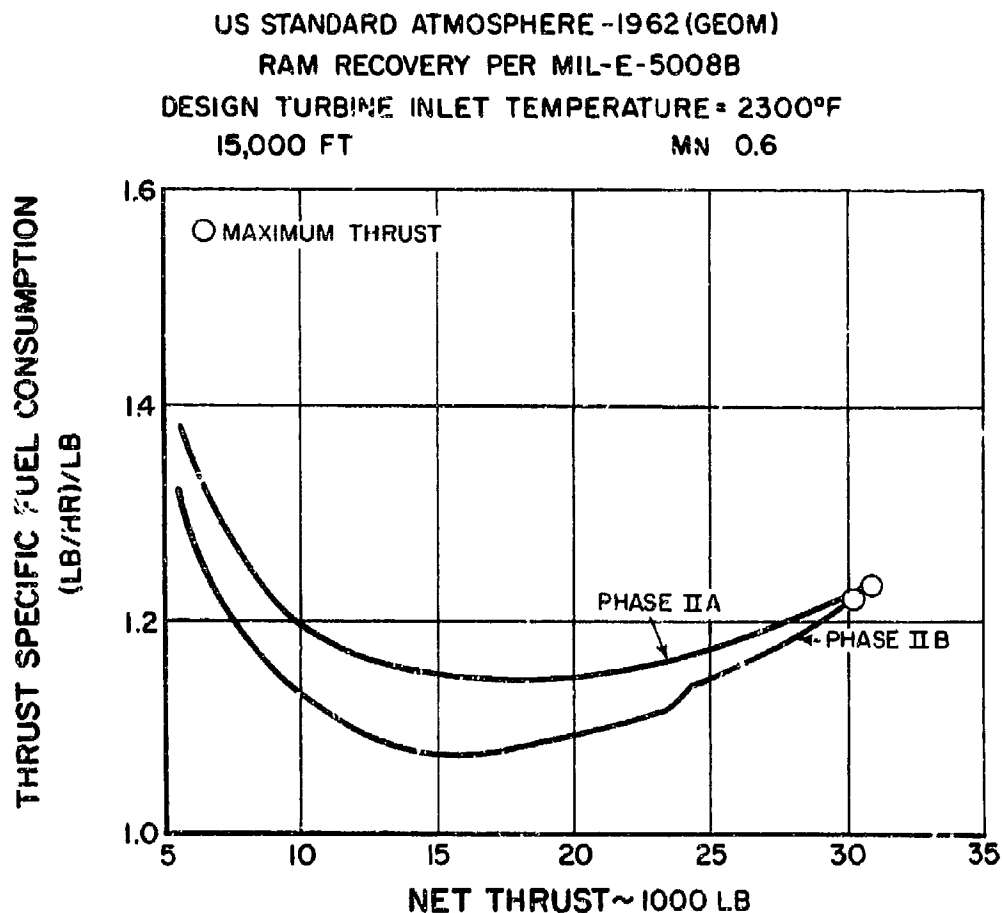
ESTIMATED PERFORMANCE OF STJ227 ENGINE AFTER PHASE IIA  
 AND AFTER PHASE IIB AT MACH 0.9 AT 36,150 FEET

Figure 1-52

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ESTIMATED PERFORMANCE OF STJ227 ENGINE AFTER PHASE IIA  
AND AFTER PHASE IIB AT MACH 0.6 AT 15,000 FEET

Figure 1-53

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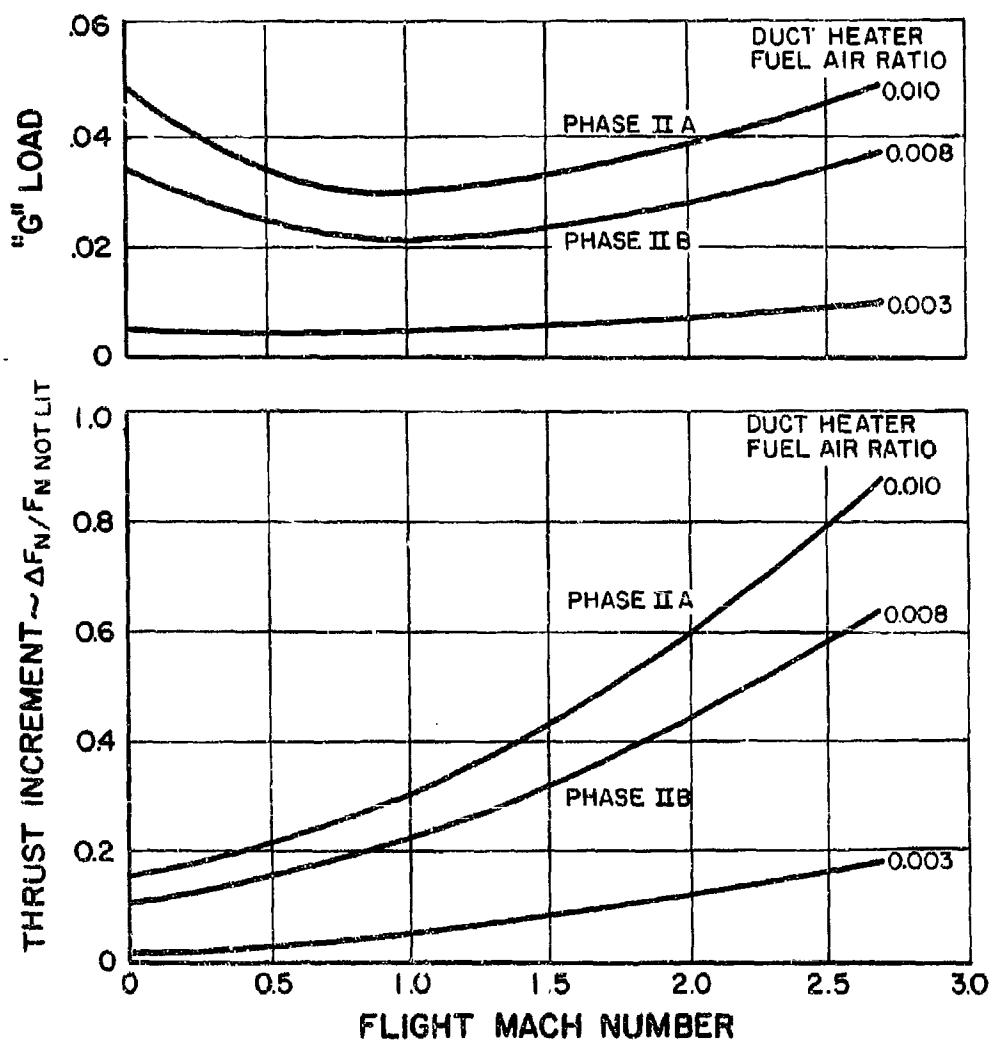
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U.S. STANDARD ATMOSPHERE-1962 (GEOM)  
 RAM RECOVERY PER MIL--E- 5008B  
 TOGW = 450,000 LBS



LOADING AND THRUST INCREMENT PRODUCED BY  
 LIGHTING DUCT HEATER OF STF219 ENGINE

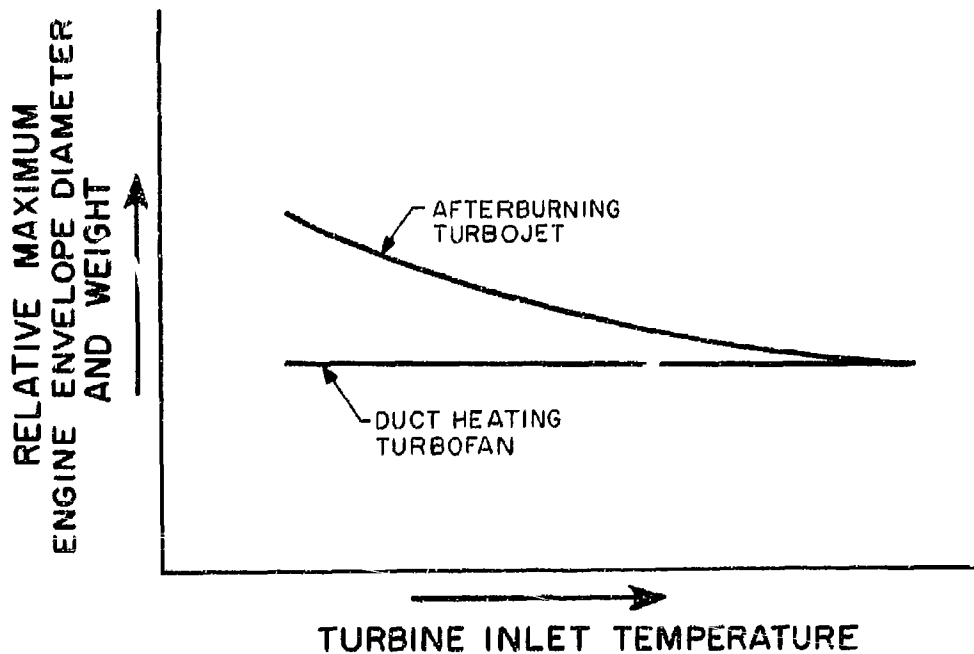
Figure 1-54

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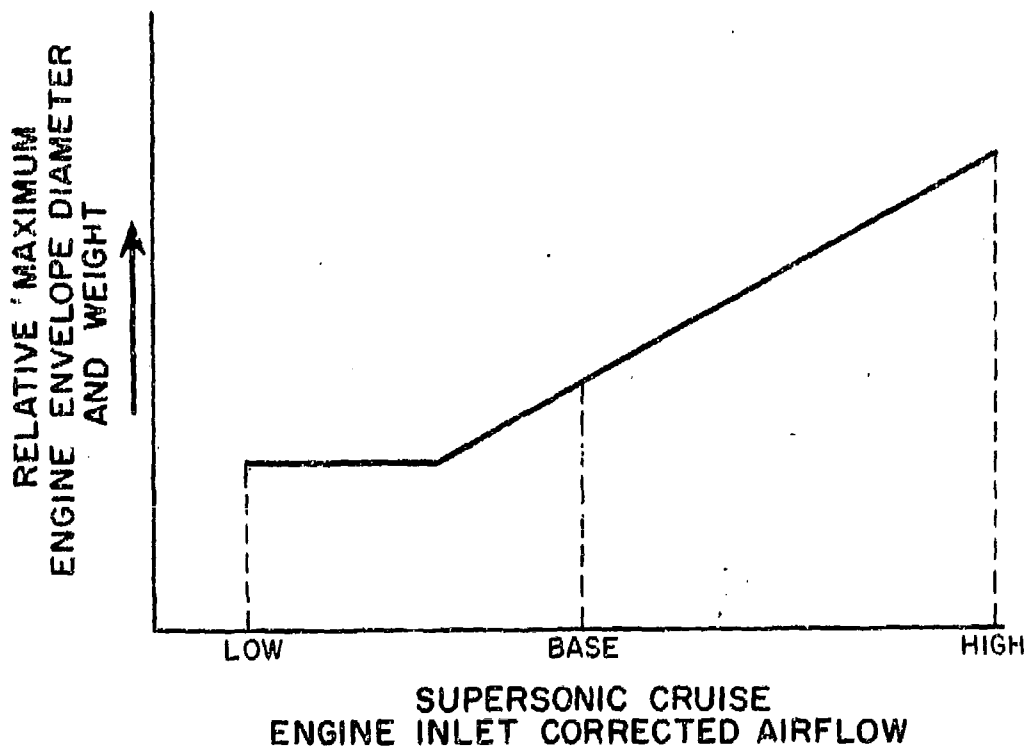


EFFECT OF TURBINE INLET TEMPERATURE ON ENGINE WEIGHT AND MAXIMUM DIAMETER

Figure 1-55

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PWA-2600



EFFECT OF ENGINE SUPERSONIC CRUISE INLET AIRFLOW  
ON ENGINE WEIGHT AND MAXIMUM DIAMETER

Figure 1-56

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